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Safe, Compact, Nuclear Propulsion

Solid Core Nuclear Propulsion Concept

October 1988

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FOREWORD

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A derivative of the NERVA nuclear rocket engine (NERVA experimental engines were successfully ground tested in the late 60's and early 70's) was evaluated for future application in Air Force Missions. This nuclear engine was compared to an advanced nuclear electric engine as well as an advanced chemical engine for use in orbital transfer, and lunar and Mars missions. The NERVA stage had advantages over the chemical stage in payload, life cycle cost, and propellant consumption. The NERVA stage also had lower mission time requirements and life cycle cost compared to the nuclear electric stage. Fabrication of the NERVA stage does not require an extensive engine development program and demonstrated engine operating characteristics assure a reliable, flexible stage with a multiple mission capability. The single propellant used in nuclear engines eliminates the combustion hazards associated with the use of dual propellants in space. Preliminary evaluations also indicated that varying the tankage used according to mission could reduce the cost of operating the nuclear engine. The open air engine tests used to develop the NERVA fuel and engine (Continued)

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19. ABSTRACT

system are not environmentally acceptable today. A proposed alternate fuel development approach is to screen candidate fuel systems and conduct failure tests in test reactors. Such testing facilities are designed for testing of closed loop systems. Failure tests on full scale core segments could also be conducted closed loop in larger facilities. A full scale engine system qualification could be accomplished but the facility must be designed to shut down immediately at the first detection of fission product release. It is estimated that all non-nuclear components of the NERVA engine system could be qualified for engine testing within four years and depending on the problems encountered, cost could range from $55 to $180 million.

EXECUTIVE SUMMARY

The Air Force Astronautical Laboratory (AFAL) has been assigned responsibility to develop a nuclear propulsion system for future Air Force applications. To that end, AFAL asked the Idaho National Engineering Laboratory (INEL) to evaluate low cost, near term concepts for a range of missions and propulsion systems.

INEL selected a team of contractors for the nuclear propulsion studies; Martin Marietta for mission analysis, Science Applications International (SAIC) for flight safety analysis, Westinghouse for the nuclear subsystem, and Rocketdyne for the engine system. As the overall team leader, INEL is responsible for engine ground testing and ground test facility design. This report comprises results from four studies performed for the AFAL by the contractors.

Section 1 contains the results of a preliminary study completed for the Air Force Astronautics Laboratory to evaluate the life cycle cost advantages of using a nuclear rocket for orbital maneuvering and transfer missions. A derivative of the NERVA nuclear rocket engine was selected for the preliminary evaluation. NERVA experimental engines were successfully ground tested in the late sixties and early seventies, and a flight qualification engine was in advanced design stages when the program was cancelled. Design documentation has been retained on this engine, and an engine could be made available for Air Force use in the mid to late 1990s.

The reference engine gave life cycle cost advantages of $7 to $19 billion over chemical engines in performing various combinations of orbital transfer and maneuvering missions projected for the years 1995 to 2020. The cost advantages resulted from the higher specific impulse of the nuclear engine (970 vs 475), the logistical advantages of handling one propellant versus two, and the versatility of the engine in performing the full gamut of orbital maneuvering and transfer missions.

The initial operating goal for a nuclear rocket engine would be to provide $\sim$10,000 to 30,000 pound thrust at high temperature (2700 to 3000 K) for a total of 10 hours, spread over $\sim$160 reactor operating periods of 2 to 8 minutes each.

Section 2 reports on studies to expand the evaluation of the nuclear engine in performing space missions. The life cycle cost of operating the engine was expanded to include a comparison with an advanced nuclear electric engine as well as the advanced chemical engine. The three engine concepts were compared in orbital transfer, lunar, and Mars missions. Sensitivity studies were conducted to determine the engine characteristics which had the greatest impact on life cycle costs, and the potential advantage of having two sizes of nuclear engines was addressed. A preliminary safety evaluation was completed.

The nuclear electric propulsion (NEP) engine selected for comparison used an electromagnetic thruster (MPD) with a 500 kWe nuclear electric system producing 7 N thrust and 4000 sec specific impulse. The engine was sized to efficiently accomplish an orbital transfer from low earth orbit (LEO) to geosynchronous orbit (GEO). In performing the LEO to GEO orbital transfer the nuclear engine had life cycle costs lower than the chemical engine at any level of mission activity greater than three missions per year over the 26 year period evaluated. The nuclear electric engine had higher life cycle costs at any level of mission activity.

In support of a Lunar base, the nuclear rocket has significant advantages, especially during the initial build-up of the base. Chemical propulsion schemes often assume Lunar propellants; however, careful analysis casts doubt upon the economics of Lunar liquid oxygen (LLOX) when compared to the Earth supplied cost of $750/lb. Without an aerobrake the nuclear stage uses half the propellant of a chemical system--even with LLOX usage. The simplicity of Earth propellants and lack of aerobrakes heavily favor use of a nuclear stage. Electric propulsion cannot compete because of its 300 day flight time which offsets its propellant savings by requiring more vehicles and higher ground operations costs. The nuclear

rocket is also an excellent candidate for a Lunar lander/ascent vehicle because contamination problems are minimal and thrust-to-weight requirements are lower.

For Mars base missions, the nuclear rocket is the only propulsion method that can support Cycling Astronautical Spaceships for Transplanetary Long-duration Excursions (CASTLEs) using only terrestrial propellants. The combination of high thrust-to-weight taxis and large delta-Vs prevent both chemical and electric propulsion from supporting cycling orbits. If direct ballistic trajectories are employed, 30 to 75% propellant savings can be realized by use of the nuclear engine. Electric propulsion is heavily penalized because its flight time is 728 days, versus a ballistic transfer of 200 days. Because of the losses associated with low thrust, a nuclear rocket with an aerobrake actually uses less propellant than the electric propulsion stage.

The sensitivity studies showed that the specific impulse of the nuclear engine was the dominant factor in the total propellant required to perform a mission. Engine weight had approximately one/half the impact of specific impulse, and other factors evaluated were relatively insensitive. Cost of getting material to orbit had the greatest impact on the stage operating costs. Engine reusability became a significant cost item if the number of missions dropped below about ten. A preliminary evaluation of a small stage for orbital maneuvering missions and a larger stage for orbital transfers from LEO to GEO indicated that reductions in the cost of operating the nuclear engine could be obtained by varying the tankage used in various missions.

A preliminary review of safety and reliability issues determined that the ability of the ANRE to produce thrust without combustion avoids many of the safety problems associated with the use of dual propellants in space. Also, the prior Rover/NERVA provides an extensive technology base and engineering practices for design and development of safe nuclear rocket engines. Further, safety policies, practices, and approval processes exist for the use of nuclear reactors in space.

The nuclear engine is inherently safer than chemical engines in orbital transfer applications. The single propellant eliminates the possibility of explosive combination of propellant and oxidizer during launch and storage, and lower component stress levels should give more reliable operation. Because of the relatively short burn times (2-8 min.) required for orbital maneuvering missions, with proper design it is feasible to plan on servicing payloads shortly after shutdown and limited manual work on critical engine components at times as short as several days after engine shutdown. Safe disposal of a malfunctioning engine is not a problem in orbital transfer applications.

Section 3, adapted from EG&G's September 1987 report "Fuel Test and Qualification Requirements", addresses the key issue of qualifying a nuclear engine for flight testing. The open air engine tests used to develop the NERVA fuel and engine system are not acceptable in today's environment. Testing a reactor without containment is not acceptable from the safety standpoint, and release of the nozzle exhaust without cleanup of fission products is not environmentally acceptable.

The fuel development approach used during the NERVA program was to test candidate fuel systems electrically and then to fabricate full cores to evaluate fuel performance in the nuclear environment. This approach resulted in the extensive release of fission products in the nozzle exhaust. It is possible to design a full scale engine test stand that could clean the nozzle exhaust to acceptable environmental standards prior to release, but the cost would be very high and the operating costs would be prohibitive.

An alternate fuel development approach is proposed for the ANRE engine. It is proposed that candidate fuel systems be screened and failure tests conducted in test reactors such as the Advanced Test Reactor (ATR) at INEL. Tests could be conducted closed cycle with hydrogen circulating through cooldown and cleanup systems prior to returning to the test element. The reactor was designed for testing of closed loop systems and

modest costs have been estimated for fabrication of the test loop and for operating costs.

Failure tests on full scale core segments could also be conducted closed loop in larger facilities such as the LOFT containment facility at INEL. This facility was designed to conduct failure tests on a scale core of a commercial pressurized water reactor, and has the capability of removing up to 60 MW of heat from within the containment without external circulation of primary loop coolant. The ability to clean up a primary system with failed fuel has been demonstrated.

With the fuel and core qualified in existing facilities, such as those at INEL, a facility to qualify a full engine system should have reasonable operating costs, i.e., it would be designed to shut down immediately upon the first detection of fission product release. It would not be possible to qualify fuel in the facility, but full scale engine system qualification could be accomplished.

In Section 3, details of testing two candidate reactor concepts being considered by AFAL are addressed, these are the NERVA concept being studied by the INEL team and the particle bed concept being studied by the Brookhaven team. Potential failure modes of the two concepts are investigated and means of studying critical areas of failure are proposed. Generally, the NERVA concept will be easier to test, because of the lower power density in the fuel elements (3.6 kW/cm^3 versus 8.2 kW/cm^3), but both systems can be tested.

Section 4 reports on studies by Rocketdyne to identify critical issues, define testing requirements, estimate costs and schedule for qualifying the non-nuclear components of the NERVA engine system. To accomplish these tasks Rocketdyne conducted preliminary systems analyses to establish component requirements and then identified concepts for all major non-nuclear components. The engine configuration was based on the NERVA design but all components were upgraded to current day technology.

No technical issues were identified which required a significant research effort for resolution. It is estimated that all non-nuclear components could be qualified for engine testing within four years and the costs could range from 55 to 180 million depending on the problems encountered. The problems which could result in the wide range of costs are delineated.

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Section 2. Advanced Nuclear Stage Mission Analysis - Task 2 Report Parametric Analysis. Work Completed by the Consortium to Develop an Advanced Nuclear Rocket Engine: Idaho National Engineering Laboratory; Martin Marietta; Rockwell International; Science Applications International Corporation; Westinghouse Electric Corporation.

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Section 4: Advanced Nuclear Rocket Engine - Preliminary Engine Studies, Published August 1987. Work completed by Rocketdyne Division, Rockwell International.

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1. BASIC NUCLEAR ENGINE EVALUATION

1.1 Introduction

The need for an advanced nuclear rocket propulsion system was identified in Project Forecast II, the Air Force Systems Command report which looks into future Air Force needs.

To fill that need an INEL team, under contract to the Air Force Astronautical Lab, has produced program plans for both the engine system and the ground test facility. The contract calls for studies to assess missions for which nuclear engines would be most effective and then to begin work to establish engine requirements. This section presents preliminary results evaluating the cost advantages for using a nuclear rocket in orbital transfer and maneuvering missions.

The results of the mission analyses by Martin Marietta (Martin) were so favorable to nuclear stages that arrangements were made for SAIC to perform an independent review of the analyses. The review by SAIC involved examination of Martin's presentation material, some independent calculations of payload performance and attendance at a briefing by Martin, with the opportunity for discussion. In general, the SAIC independent audit showed substantial agreement with Martin's results - see Appendix D.

A derivative of the NERVA nuclear rocket engine was selected for the preliminary analysis. Technology for this engine concept is fully developed and it represents the earliest nuclear engine the Air Force could place into operational use. Westinghouse, the nuclear subsystem contractor for the NERVA engine, has retained a complete set of all drawings, specs, and procedures for this subsystem. Rocketdyne has reviewed the non-nuclear components and has identified improved currently available components in all areas. A review of operating requirements indicates the stress levels necessary for operating the nuclear engine non-nuclear components are well below current design requirements for similar chemical engines. These findings and the stringent design practices used on the original NERVA engine should result in the rapid development of a highly reliable engine.

An engine thrust of 14,550 lb with a weight of 4600 lb and a specific impulse of 970 sec was estimated for the reference engine. This estimate was based on upgrading the gamma engine designed by Los Alamos at the close of the NERVA program. The upgrades were based on low risk technologies which have been developed since the NERVA program was terminated in the early 1970s. Requirements were established that the engine would be operated for the equivalent of 10 full-power hours and that it would be used for 80 missions. These requirements were based on tests conducted during the development of the engine.

Orbital transfer and maneuvering missions were selected for the preliminary evaluation of the cost advantages of the nuclear engine. Orbital missions were selected for this analysis because the advantage for the high specific impulse is maximized in this application, the disadvantage of the low thrust to weight is minimized, and the nuclear engine has safety advantages.

The space transportation architecture study (STAS) mission level III-3 "partial SDI deployment" was selected for the preliminary life cycle cost analysis. This is an intermediate level of mission activity for both NASA and DOD. The STAS models are frequently updated and initial studies were conducted using the model current as of January 1986. A follow-on study was conducted using the STAS model current as of December 1986.

For the life cycle cost comparison, all rules of the STAS studies were used for boosters and for propellant in orbit costs. For the chemical engines, all information was taken from the NASA OTV and OMV studies. For the nuclear engine, it was necessary to estimate cost. For DDT&E costs $2 billion were used and $30 million for engine cost. The nuclear engine was operated from a space platform and its life cycle costs included the cost for two space platforms: one at a high inclination and the second at a low inclination. An additional 6% was added to the propellant requirements for the nuclear engine to account for any propellant required for cooldown. This is a conservative assumption, as most of this propellant would be used to produce useful thrust.

For the first study, the costs for conducting 1312 orbital maneuvering missions and 230 orbital transfer missions in the time period 2010-2020 with nuclear and chemical engines were compared. The cost for the chemical engines was available from the NASA data for a direct comparison of the engines. The nuclear rocket engine gave a 6.9 billion 20% cost saving over the chemical engines.

A second study was conducted by adding missions in which the propulsion system was integral with the stage. In this case it was not possible to directly compare the nuclear and chemical systems, since the costs for the propulsion system could not be separated from other stage costs for the integral missions. To conduct the comparison, the conservative assumption was made of comparing the total costs of the nuclear engine (propellant and engine) with only the propellant costs for the chemical engine. The time period was increased to 1995-2020, and in this case the advantage for the nuclear engine increased to $18.6 billion. This study was repeated using the December 1986 STAS model and the advantage dropped to $15 billion due to a lower total number of missions (6655 vs. 10456 payloads).

The primary advantage for the nuclear engine is the reduced propellant requirements due to the higher specific impulse. The use of a single propellant also results in logistical and operational advantages which have been costed in the model. There are, however, many additional intangible advantages which are difficult to put in a cost model and would increase the true advantage of the nuclear engine. The radiation environment will make it necessary to delay operations near the engine for several days unless protection is provided for workers. This will increase costs but should be manageable since the short burns required for orbital operations result in a relatively low level of radiation. In the area of the payloads a minimum of shielding would permit access as soon as needed.

The nuclear engine is inherently safer than the chemical engines. During launch the core is not radioactive and any launch accident would result in the return of harmless fuel to earth. Water immersion of a full core will not result in criticality. The oxidizer and fuel of the chemical

engines can explosively combine at any time during launch, storage or engine use. The single nuclear engine propellant is less of a problem during launch, and does not have an explosion problem during storage or use. By limiting the nuclear engine to short burns while in low earth orbit and establishing proper design criteria for the nuclear engine, it can be designed to avoid serious core disintegration after a loss of coolant accident from full power. This will protect the stage and permit subsequent removal of the engine to a safe orbit.

1.2 Baseline Engine

The baseline selected for the mission performance analysis engine is a NERVA-derivative engine with a full power design thrust level of 14,550 lb, as compared to the 75,000 lb thrust of the NERVA engine design when the NERVA program was terminated in 1972. The size of the baseline engine has been reduced, compared to the NERVA engine, to account for this change in the full-power thrust level. During the 25 years since termination of the NERVA program, there have been numerous space technology advances that are applicable to improvements in the basic NERVA engine design that existed in 1972. For the preliminary mission analysis reported herein there has been no attempt to redesign the engine to take full advantage of such improvements. However, adjustments to the baseline ANRE mass have been made to account for low risk technology improvements. Also, because of significant improvements in nuclear fuel technology that permit higher operating temperatures, the specific impulse of the baseline engine has been increased to 970 s, as compared to 825 s for the NERVA engine. The specific impulse of the baseline ANRE is more than twice that of comparable chemical engines.

Compared to chemical rocket engines, the NERVA-Derivative engine has extensive operational flexibility. As discussed further in Appendix A, the NERVA type engine has the capability to readily and efficiently (at high Isp) operate over a relatively wide range of thrust output. Because of this feature it is feasible to satisfy the propulsion requirements of a wide spectrum of space missions with engines of a single size.

The thrust level (14,550 lb) of the baseline ANRE utilized for the mission analyses reported in Appendix B is not based on optimization studies. It is considered to be in the range of sizes that would be attractive for many of the currently projected Earth orbital missions and is about the size of some NERVA-type engines that have been investigated, e.g., the SNRE. Information from the latter has provided applicable information relative to the projected engine weight and performance.

Appendix A provides additional descriptive material about the NERVA-type engine, its principal components, and operating characteristics.

1.3 Operational Concepts for Nuclear Stages

Nuclear stages based on LEP (Low Earth Platforms) are ideally suited for space operations. In our scenarios, these stages collect single payloads or multiple payload pallets from 150 nm orbits and deliver them to LEP, GEO (Geosynchronous Earth Orbit) or multiple-GEO destinations or beyond. Nuclear stages will also be able to retrieve payloads from multiple GEO (or lower) altitudes, deliver them to space platforms for maintenance/repair or to a Shuttle for return to Earth. We believe all of these missions can be flown with one basic nuclear stage using different sizes of propellant tanks.

The principal logistical support requirement for nuclear stages involves periodic resupply (LH_2 propellant and spare parts) of two space platforms (one each for high and low inclination orbits) by "space available" and/or dedicated tanker/cargo vehicles. Scenarios for LEO, GEO, and multiple-GEO missions are illustrated in Figures 1 and 2, below. Because of the similarity of these profiles, it is expected that only one avionics package (GC&N and ACS) will be required for such missions.

1.4 Operational Concepts for Chemical Stages

Unlike nuclear stages (always space-based), chemical stages can be either ground or space-based. The categories of ground based chemical stages include:

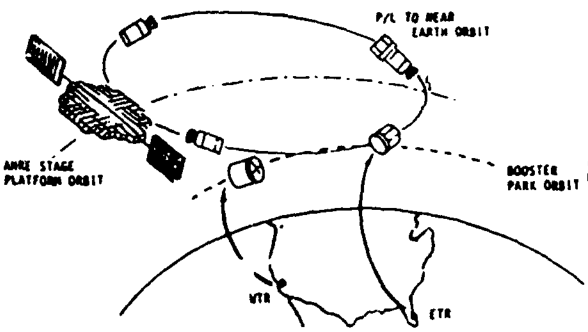

Figure 1. Integral and tug orbits.

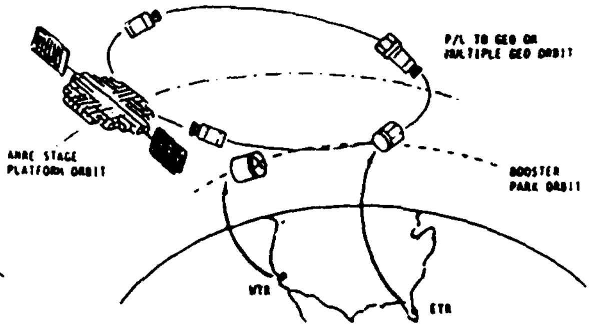

Figure 2. GEO/Multiple GEO orbits.

o Integral Stages (those with payloads small enough to achieve destination orbits using built-in propulsion systems). These comprise a large percentage of all payloads in most mission models.

o GB OMVs (Ground Based Orbiting Maneuver Vehicles). OMVs are stages which fly medium to heavy payloads to LEO destinations.

o GB OTVs (Ground Based Orbit Transfer Vehicles). OTVs are transfer systems which deliver payloads to LEO, GEO and high orbits.

Space based chemical stages include:

o SB OMVs (Space Based Orbiting Maneuvering Vehicles). These OMVs transport payloads from low parking orbits to LEO destinations, retrieve them for maintenance/repair on LEO space platforms, or take them to a Shuttle for return to Earth.

o SB OTVs (Space Based Orbit Transfer Vehicles). These vehicles transfer payloads from parking orbits to LEO, GEO, and Multiple-GEO destinations.

GB OMVs and GB OTVs are launched with multiple payloads (depending on payload weights and destinations) and are usually expendable. In our scenarios (see Figure 3), SB OMVs and SB OTMs would be based on a space platform, would pick up single or multiple payloads from 150 nm parking orbits, but would not be expendable.

1.5 Space Transportation Architecture Study (STAS) Mission Models

In order to design/size a nuclear stage and compare Life Cycle Costs (LCCs) for nuclear and chemical stages, it is necessary to develop mission models. As of January 1986, the joint NASA/DOD STAS program had defined five possible levels of civil and military launch activity (called "Options"). These Options are shown in Figure 4 and consist of

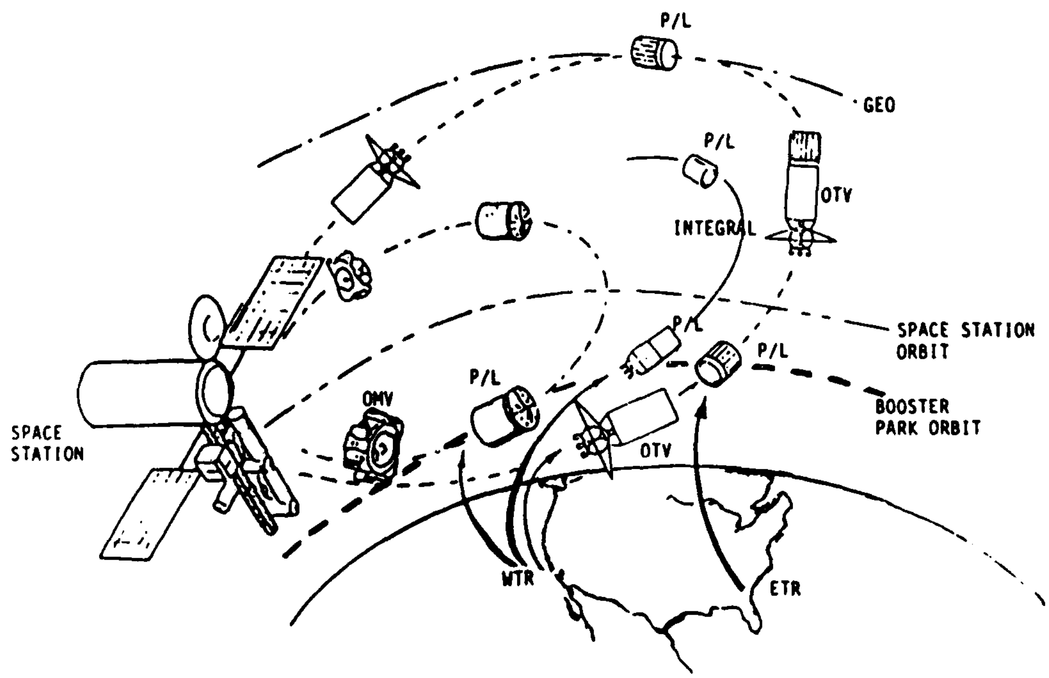

Figure 3. The storable, solid, cryogenic scenario.

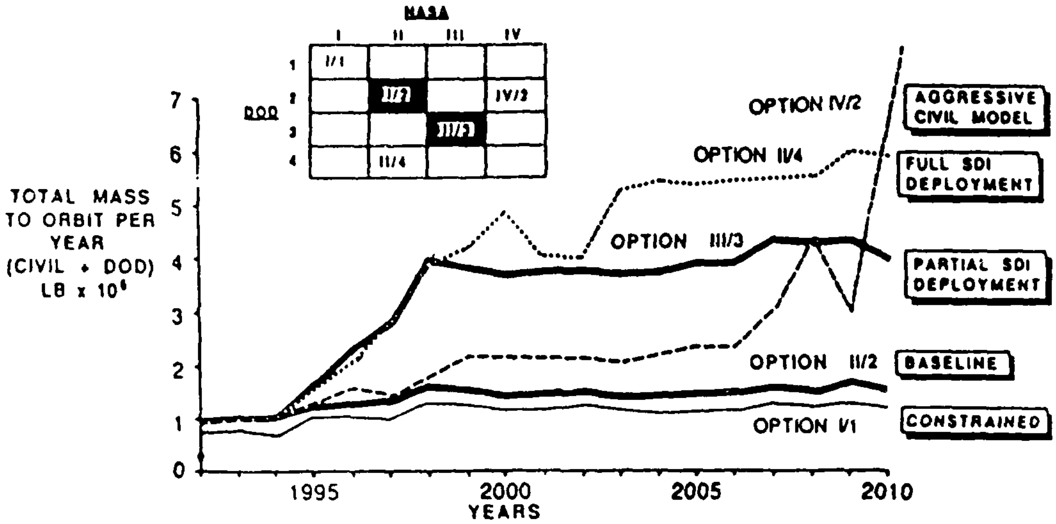

Figure 4. Five STAS NASA/DOD mission options (January 1986).

combinations of Civil Models (I, II, III, and IV) and Military Models (1, 2, 3, and 4). We selected Option III/3, "Partial SDI Deployment," as the most suitable baseline for our initial LCC analyses.

STAS NASA/DOD models are frequently updated, and in June 1986, the SDI component of Military Model 3 (Partial SDI Deployment) was reduced substantially. Figure 5 shows the "old" SDI profile (January 1986), the "new" SDI program (June 1986) and the resulting "new" Military Model 3 (December 1986). Because of potential interest in purely military applications of the nuclear stage, we also selected the "new" DOD Model 3 for an additional LCC comparative analysis.

The principal parameters in the LCC mission model include: the category of each stage (integral, OMV or OTV), number of payloads, payload weight and "delta" velocity required for delivery. These elements are shown in Table 1 for three LCC Mission Models (A, B, and C).

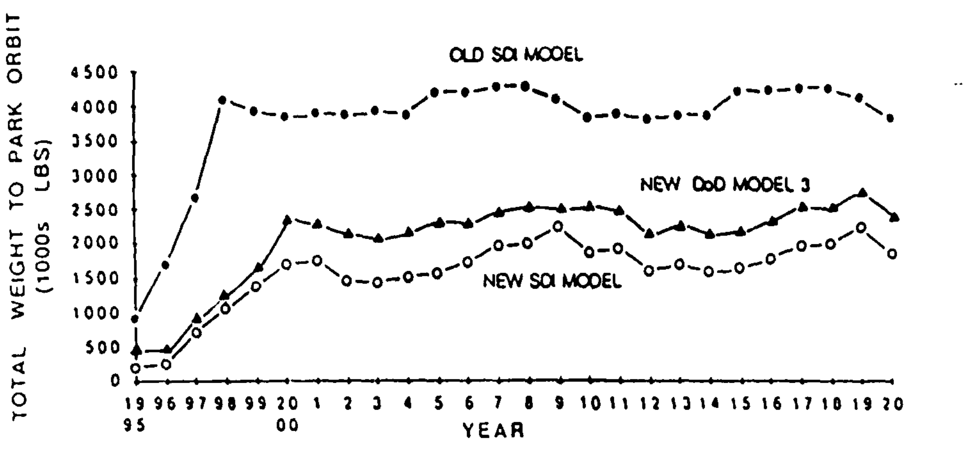

Figure 5. Old and new SDI/DOD mission models, STAS Option III/3.

Although not included in the cost analysis, the planetary missions (used in our performance and design analyses of the nuclear stage) are also shown in Table 1.

In order to calculate accurate costs and cost sensitivities, it is necessary to consider a number of mission models, each of which has been manifested in detail. Manifesting is a demanding and costly process which determines exactly how many and what type of launch events are matched with one or more payload events. Because of the complexity and cost of these manifesting operations, it has not been possible to calculate the number of launch events needed for each of the Mission Models selected for cost analysis, and we have had to make assumptions.

Figure 6 illustrates the manifesting rules used with Cat 1 (Integral), Cat 2 (OMV), and Cat 3 (OTV) payloads. By definition, the Cat 1 payloads have individual, integrated propulsion systems and are locked into a one-PL-per-stage manifesting rule. An alternative arrangement is to bundle six of these small Cat 1 payloads into a "pallet" package, boost these packages into parking orbits, and use chemical or nuclear OMVs to deliver

TABLE 1. MISSION MODELS USED FOR COSTING AND PLANETARY STUDIES

Mission	Payloads ($)	Weight (lb)	ANRE Delta Velocity (fps)	Burn Time (min)
Mission Model A (Option III/Old 3)				
Category 1 (Integral)	5,144	10,000	3,600 (LEO)	8
Category 2 (OMV GB/SB)	1,312	2000-5000	800-3600 (LEO)	2-8
Category 3 (OTV GB/SB)	230	25,000	13,900 (GEO)	20

Time Period: 2010-2020 $ Category 1 missions were excluded from this LCC analysis

Total Payloads: 1542 because reliable cost information on Integral Stages was not available.

Mission	Payloads ($)	Weight (lb)	ANRE Delta Velocity (fps)	Burn Time (min)
Mission Model B (Option III/Old 3)				
Category 1 (Integral)	8,144	10,000	3,600 (LEO)	8
Category 2 (OMV GB/SB)	1,962	2000-5000	800-3600 (LEO)	2-8
Category 3 (OTV G8/SB)	350	25,000	13,900 (GEO)	20
Time Period:	1995-2020			
Total Payloads:	10,456			
Mission Model C (Option III/New 3)				
Category 1 (Integral)	4,418	10,000	3,600 (LEO)	8
Category 2 (OMV GB/SB)	1,918	2,000	800-3600 (LEO)	2-8
Category 3 (OTV GB/SB)	319	25,000	13,900 (GEO)	20
Time Period:	1995-2020			
Total Payloads:	6,655			
Planetary (Jan 86)				
Mars (1)	TYP	35,000	16,770	8-26
Mars (2)	TYP	15,500	16,770	5-15
Galileo (1)	1	5,600	20,979	14
Galileo (2)	1	7,040	14,212	9
Pluto Orbiter	1	50,000	24,400	80
Lunar Orbiter	1	93,000	10,500	44
Neptune	1	9,600	23,800	22

These planetary missions have not been included in the LCC analysis because detailed chemical stage designs were not available for comparison.

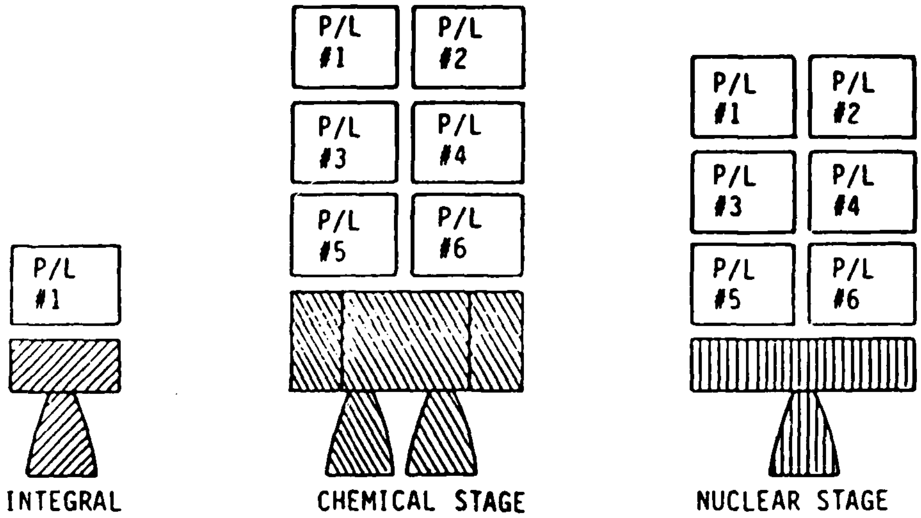

Figure 6. Examples of the two manifesting rules used.

these PLs to six destination orbits. This leads to a manifesting rule of six PLs per stage (see configurations at center and right, Figure 6), a reasonable number based on STAS manifesting experience.

This Cat 1 manifesting alternative has the advantage of (a) eliminating the need to develop, test, acquire and integrate Cat 1 propulsion systems with the Cat 1 payloads; and (b) reducing the total Cat 1 weight which needs to be placed in parking orbits. Since Cat 1 payloads usually comprise 60 to 80% of the payloads in most mission models, significant leverage is provided in reducing LCCs with this approach.

1.6 Results of the Comparative LCC Analysis

The results of the comparative LCC analysis using Mission Models A, B, and C are summarized in Table 2, below.

TABLE 2. RELATIVE LCC OF CHEMICAL AND NUCLEAR STAGES

	Billion ($)		
Mission Model A [STAS Options III/3 (Jan 86)]	Space Based Chemical Stages	Space Based Nuclear Stages	Nuclear Stage Savings
Time period: 2010-2020 (11 years) Total payloads: 1542 Total launches: 1542 Missions: OMV & OTV (no Integral)	34.1	27.0	6.9 (20.2%)
Mission Model B [STAS Options III/3 (Jan 86)]			
Time period: 1995-2020 (26 years) Total payloads: 10,456 Total launches: 3,669 Missions: Integral + OMV + OTV	123.7	105.1	18.6 (15.0%)
Mission Model C [DOD Option 3 (Dec 86)]			
Time period: 1995-2020 (26 years) Total payloads: 6,655 Total launches: 2,973 Missions: Integral + OMV + OTV	87.8	72.8	15.0 (17.1%)

The following comments are provided on our comparative costing results. See Appendix B for a more detailed listing of cost ground rules and assumptions.

Mission Model A

This model was the smallest in terms of payload events (1542). None of the 5144 Cat 1 payloads was included because reliable cost data were not

available on these one-of-a-kind vehicles. The manifesting rule used was one payload/stage for Cat 2 and Cat 3 payloads. Both chemical and nuclear stages were space-based. The operational scenario assumed these payloads were delivered to their parking orbits by ground based boosters (costed). Chemical and nuclear stages then transported the payloads to destination orbits.

Mission Model B

This model was the largest (10,456 payload events) and included all three categories of stages. All payloads were delivered to parking orbits and then transported to destination orbits by space based chemical and nuclear stages. Cat 1 payloads were manifested six-PL/booster to parking orbit and six PL/stage to destinations. The corresponding manifesting for Cat 2 and Cat 3 PLs was one-to-one. Launch costs to intermediate orbits are included in the LCC analysis as well as the cost of two platforms for the nuclear system. It was assumed that the chemical stages would operate from a Space Station.

Mission Model C

Mission Model C was an intermediate size model (6686 payload events). The ground rules and assumptions were the same as for Mission Model B.

1.7 Operational Performance of the Nuclear Stage

The unique characteristics of nuclear rocket engines give nuclear stages a decisive performance and reliability advantage over chemical stages. Figure 7 shows two demanding LEO-GEO-LEO missions: (a) deliver PL and return empty and (b) deliver and retrieve PL. The propellant loading is substantially less for the nuclear stage in spite of its heavier engine and avionics shielding. This performance advantage is even more pronounced for even more demanding missions such as interplanetary trips. This is due to the high Isp (970 s) of the reactor engine and the fact that each nuclear stage carries only one tank of propellant.

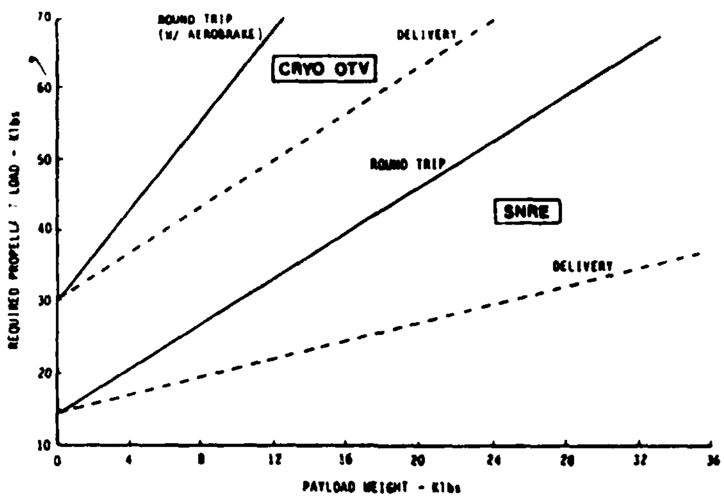

Figure 7. Comparison of nuclear/chemical stage performance for two LEO-GEO-LEO Missions.

The nuclear stage's high reliability is a result of operating conditions (for turbines, pumps, and pressurized components) that are substantially below maximum performance limits. Contrast this with cryogenic chemical engines that use advanced materials and push operating limits to maximize specific impulse (Isp) output.

The inherent simplicity of the nuclear stage also provides significant operational advantages. Since it uses only one propellant, transportation, storage and handling problems are reduced.

1.8 Nuclear Stage Design

Our design of a nuclear stage was accomplished with conceptual level models. Delta velocity requirements for various missions were derived using the ideal rocket equation and checked against STAS data (see Table 1).

The engine used was a modified gamma version of the SNRE (small nuclear reactor engine) employing current materials. No consideration was

given to ullage, residuals or propellant boil-off. A conservative tankage design fraction (0.075) was used to ensure withstanding GB launch acceleration loads. The lump sum weight of 500 lb allocated to miscellaneous components is consistent with chemical OTV design practice and includes a growth margin of approximately 100 lb.

The ACS, thermal control, avionics, and pressurization/feed systems were estimated at 811 lb. We added 100 lb extra for shielding of avionics. The ACS uses warm gas thrusters from accumulator tanks. Hydrogen gas, transformed from a liquid state during the reactor operation and cooling cycles, is used to recharge these tanks. It was assumed that 2% of the total propellant is diverted to the ACS function. Propellant loss due to reactor cooling was estimated to be no more than 6% (assuming no thrust during the cooling cycle). A summary of the design ground rules and assumptions is provided in Table 3.

TABLE 3. GROUND RULES AND ASSUMPTIONS FOR NUCLEAR STAGE DESIGN

Sizing	Weight (lb)
Item	
Engine dry weight	4,600
Pressurization and feed systems	200
Avionics	400
External shielding (components)	100
Thermal control	150
Altitude control	61
Misc: P/L adapter; structural attachments, etc.	500
Total dry weight less tankage	6,011
Tankage: Used 0.075 x Wp (weight of stage propellent)	
ACS Propellant: Use H_2; assume 2% of total Wp	
Performance	
Isp: 970 s	
Thrust: 14,550 lb	
Flowrate: 15.0 lb/s	

With the stage weights for each mission known, we could determine vehicle size from consideration of constraints on vehicle size, the fuel density and desired layout. The dimensional limits included a maximum diameter of 14.5 ft (Shuttle cargo bay constraint). Figure 8 shows a longitudinal cross-section of the nuclear stage, including the toroidal tanks which provide emergency LH_2 for cooling in the event of a failure of the primary cooling system.

1.9 Safety Considerations

The baseline ANRE, a NERVA derivative, utilizes a single propellant (hydrogen) and therefore is inherently safer than chemical rocket engines, which require dual propellants. The fuel and oxidizer of chemical rockets have the potential for unplanned combination and explosive ignition at any time during launch operations, space storage, or mission use. This possibility does not exist with the NERVA type engine.

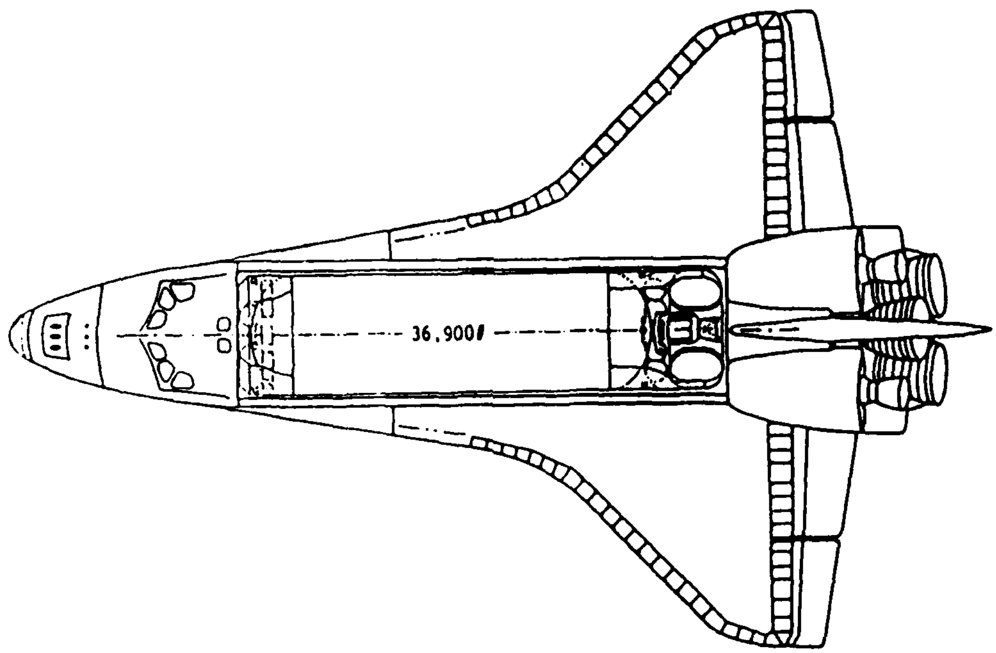

Figure 8. Layout of a typical nuclear stage within the shuttle bay.

The objective of the NERVA program was the development of a nuclear engine suitable for manned missions to Mars and consequently the reliability and safety of the engine design were of paramount concern during all phases of the program. It follows that a major and high priority effort was directed toward eliminating from the engine design those single failures or credible combinations of errors and failures which could endanger mission completion, the flight crew, the launch crew, or the general public. Probabilistic design and failure mode and effects analysis were included in this effort. Examples of the effects of these analyses on flight engine design were the incorporation of redundant turbopumps and the use of four valves in place of each single valve. Where no practical engine design solutions were found for credible single or multiple failures that could jeopardize crew or population safety, appropriate countermeasures and alternative operating modes were explored. For example, provisions were made for engine operation in an emergency mode to effect safe crew return and to prevent danger to the Earth's population in the event the planned mission had to be abandoned because of an engine failure.

Compared to the manned missions to Mars, the "near" earth space operations of current primary interest involve performance requirements that are considerably less demanding. First, the thrust (and thus engine power) required is of the order of a fifth or less than that of the NERVA engine. Second, the orbital maneuvering operations, which constitute the majority of the missions applicable to space-based nuclear stages, require burn times in the range of 3-8 minutes. For these relatively short burn times, the engine does not become highly radioactive and limited manual servicing of the engine system components above the core shadow shield is practical. Most of the engine system operating components (e.g., pumps, valves, actuators) are in this region. For example, the projected radiation levels in this area following a 5-minute burn are 0.5 rad/hr and 0.03 rad/hr for 1 and 10 days after shutdown, respectively. Thus, individuals could provide about 10 hr and 150 hr, respectively, of manual service in this area without exceeding the present guideline limit of 5 rads/year for radiation workers.

Starting with the United Nations (UN), governmental organizations have made extensive provisions to permit the effective utilization of nuclear systems in space while ensuring such use is accomplished safely. The U.S. policy, guidelines, and the review and approval process are summarized in Appendix C.

Although the current system and documentation of safety criteria and guidelines were not in existence at that time, most of the provisions in the applicable safety criteria and guidelines were implemented in the NERVA development program. Thus, high confidence exists that all space nuclear safety requirements can be satisfied by a NERVA-Derivative type engine designed for earth orbital operations.

Significant points concerning the safety of nuclear powered rockets are summarized below.

- The use of nuclear reactors in space is accepted and provided for by U.S. and U.N. policies.
- Safety specifications and criteria exist for reactors used in space.
- Design practices exist, based on Rover/NERVA experience, for safely designing nuclear rockets.
- A safety technology base supported by extensive experiments has been developed.
- A safety approval process is in place.

1.10 Conclusions

The ability to produce thrust without combustion, the resultant simplification of hardware and enhanced engine reliability, operationally safe levels of engine radiation, and dramatically higher specific

impulse (Isp) all make the Advanced Nuclear Rocket Engine (ANRE) ideal for space-based propulsion.

In addition, our analyses have shown that one ANRE engine design will be capable of most space transportation missions with significant LCC advantages over chemically powered engines.

1.10.1 Nuclear Stage Advantages

Space-based nuclear stages have a substantial LCC cost advantage over conventional space based and ground based stages for all missions in the LEO, MEO, GEO, multiple GEO regions and beyond.

Although weighing more than a chemical stage of equal thrust, nuclear stages have a clear performance advantage due to Isp of at least twice that of chemical stages.

Nuclear stages are inherently more reliable and safer than chemical stages due to:

- Simplicity of design (fewer tanks, lines, turbines and pumps)
- Less stress on systems/subsystems (larger margins between maximum and operating limits for temperature and pressure)
- Nuclear stages pose no explosive danger to payloads, platforms, stations or other vehicles, since there is only one propellant
- Nuclear stages impose no threat to ground based launch facilities during initial launch since unused reactor elements have negligible radioactivity
- Operational flexibility and adaptability to a wide range of new space based applications, e.g., ad hoc space reuse operations
- Superior operational durability (80 missions)

- Nuclear engine can be designed to generate power in addition to propulsion

- Obsolete nuclear stages can be safely disposed of in space with available residual fuel.

1.10.2 Nuclear Stage Disadvantages

Adverse public reaction to flying/launching/orbiting anything with "nuclear" components may exist.

It is difficult to inspect nuclear stages directly or to return them to earth for physical examination.

A space-based nuclear stage system requires a dedicated platform at a low inclination because it cannot be based at the Space Station.

2. ADVANCED NUCLEAR STAGE MISSION: PARAMETRIC ANALYSIS

The first task under the AFAL contract involved the preliminary evaluation of the potential life cycle cost (LCC) advantages of utilizing a nuclear rocket engine for performing Air Force space missions. For those studies, the Advanced Nuclear Rocket Engine (ANRE), a scaled down NERVA derivative, was used as the baseline nuclear engine to compare against chemical engines for performance of orbital transfer and maneuvering missions. The results of those Task I studies are reported in Section 1 of this document.

The conclusions reported in Section 1 from Task I identified a number of advantages and disadvantages of the nuclear stage. The space-based nuclear stage was found to have a substantial LCC cost advantage over conventional space-based and ground-based stages for all missions in the LEO, MEO, GEO, and multi-GEO regions and beyond. Although weighing more than a chemical stage of equal thrust, nuclear stages were identified to have a clear performance advantage due to a specific impulse of at least twice that of chemical stages (970 vs. 475). Nuclear stages were evaluated to be inherently more reliable and safer than chemical stages. Other advantages include: greater operational flexibility and adaptability to a wide range of new space-based applications, for example, ad hoc space rescue operations; superior operational durability; and the potential ability to generate power in addition to propulsion. Disadvantages consist of: the adverse public reaction to launching, flying and orbiting anything with nuclear components; the difficulty in inspecting nuclear stages directly or in returning nuclear stages to earth for physical examination; and the requirement for a dedicated orbiting platform for the space-based nuclear stage because of the undesirability of basing nuclear stages at a manned space station.

This section provides the results of the Task 2 studies under the AFAL contract. The objectives of the Task 2 studies are to extend and augment the preliminary studies of Task 1 in order to enhance the assessment of the relative merits of a nuclear rocket engine for Air Force needs. Emphasis is placed on parametric analysis of the ANRE stage, development of

operational scenarios to support potential needs and comparison of operational and cost characteristics with other propulsion systems. Accordingly, this report includes:

- ANRE stage designs for the Low Earth Orbit (LEO) to geosynchronous orbit (GEO), Lunar base support, and Mars base support missions

- ANRE design implications for DOD missions

- Comparisons between the use of nuclear, cryogenic, and electrical propulsion for each of the three missions

- An assessment of performance and manifest sensitivity as a function of reactor stage component reuse variations, for 20, 40, 60, 80, 160, and 200 missions

- Parametric analysis, for the LEO to GEO ANRE stage design, of weight, specific impulse, propellant loss due to cooldown, and unit and development costs

- The effect on LCC of variations in the costs of getting propellant to orbit

- Discussions of safety issues affecting use of nuclear stages.

2.1 Background

The nuclear rocket engine that is the central focus of the studies reported herein is the same engine that formed the basis for the preliminary studies reported in Section 1. It is a NERVA derivative, termed the Advanced Nuclear Rocket Engine (ANRE), with a specific impulse of 970 sec, nominal thrust of 15,000 lb, and weight of 4600 lb. Its principal features and developmental history, as well as the reasons for its selection, are described in Section 1.

The Task 1 preliminary studies were limited to comparing the ANRE to chemical engines for Earth orbital transfer and maneuvering operations. The more extensive studies reported in this section include comparisons with electric propulsion and extend the missions analyzed to those involved in support of Lunar and Mars bases. The emphasis remains on the Earth orbital transfer missions of primary interest to the Air Force and these are the only missions for which life cycle cost analyses are conducted. Comparisons for the Lunar base and Mars base support are based primarily on operational characteristics of the competing stages with particular emphasis on propellant requirements inasmuch as stage operational costs relate strongly to propellant demand.

Table 4 provides some general operational characteristics for the three engine technologies (chemical, nuclear, electric) that are the focus of this report. Some general observations are provided with respect to the three competing engine technologies. Chemical engines are highly developed, have been the backbone of space propulsion thus far, and will continue to serve a major role in space transportation. Neither nuclear nor electric propulsion systems have been fully developed to serve space propulsion needs. The NERVA nuclear rocket program conducted in the sixties and early seventies provides a substantial technology base for the ANRE nuclear engine. A brief review of electric propulsion technology is provided in Appendix E. An electric propulsion engine for application to the types of missions analyzed in this report will require a nuclear power source; thus, the term nuclear-electric-propulsion (NEP) is applicable. Both the nuclear and NEP engines will require extensive development. Because of the long transfer times associated with the NEP engine, it may not satisfy many of the future space transportation needs. Therefore, it is not a "stand alone" system; it requires that either chemical or nuclear engines also be available. The higher specific impulse, lower propellant requirements, and throttling potential of the nuclear engine make it a strong candidate to serve the majority of century-21 space propulsion needs, with chemical and/or NEP engine support in those special circumstances where their unique characteristics are advantageous.

TABLE 4. OPERATIONAL CHARACTERISTICS OF THREE ENGINE TECHNOLOGIES

<u>CHEMICAL</u>

Low Specific Impulse
High Thrust to Weight
Bi Propellant
Lunar Based Oxygen Generation Alternative
Compatible With Aerobrakes
Impulsive Transfers
Flight Times
- LEO-GEO <1 day
- LEO-Lunar Base $\sim$3 days
- LEO-Mars Base $\sim$200 days

<u>NUCLEAR</u>

Medium Specific Impulse
Medium Thrust To Weight
Mono Propellant
Earth Based Propellant (LH_2)
Compatible With Aerobrakes (Controversial)
Impulsive Transfers
Flight Times
- LEO-GEO <1 day
- LEO-Lunar Base $\sim$3 days
- LEO-Mars Base $\leq$200 Days

<u>ELECTRIC</u>

High Specific Impulse
Low Thrust To Weight
Mono Propellant
Earth Based Propellant (Ar)
No Aerobrake
Spiral Transfers
Flight Times
- LEO-GEO >50 days
- LEO-Lunar Base >300 days
- LEO-Mars Base $\sim$2 years

As noted earlier, the studies reported herein placed the primary emphasis on the analysis of earth orbital missions of interest to the Air Force. Based on the STAS Civil and DOD mission models, a nuclear stage capable of delivering 14,000 lb from LEO to GEO and return was defined. This stage, which accommodates 100% of DOD Options 2 and 3 and 98% of Civil Option II, formed the basis for parametric analysis of nuclear stage

performance, nuclear stage operations cost sensitivity studies, and LCC analysis of the nuclear stage. A chemical stage and an NEP stage (see Appendix E) were also defined to support the LCC analysis.

The results of the parametric analysis of the LEO-GEO nuclear stage are portrayed by Figures 9 and 10, which show, respectively, the relative impact of performance parameters on total mission propellant and vehicle dry weight. Specific impulse has by far the greatest impact, with stage dry weight quite important to total propellant required. The propellant tank fraction is next in importance in impact on both total propellant and dry weight. The shutdown cooling propellant has a noticeable impact, but the impact of both propellant boiloff and velocity loss are quite negligible.

Summary results of the LEO-GEO nuclear stage operations cost sensitivity analyses for 20 flights are provided in Table 5. This table is based on changes from the baseline value of each parameter. The table indicates the strong sensitivity of stage operations costs to changes in reliability, payload-to-orbit cost, and percent propellent that is scavenged in space from prior operations. The sensitivity of operations costs to changes from the baseline values of engine life and structure life is low. However, the values in the table do not apply to large decreases from the baseline values of 80 flights for engine life and 100 flights for structure life. The low sensitivity indicates there is little incentive to strive for increases in either engine or structure life beyond the baseline values.

Some numerical results of the LCC analysis of the LEO-GEO missions are presented in Table 6. The bottom three sets of costs in the table correspond to the total missions in each of the three mission models described in Section 1. Comparing LCCs for the nuclear and electric electric propulsion stages (first and last columns), nuclear has a relatively small cost advantage over electric propulsion. Considering that the long transfer time if the NEP stage would obviate its use for some, if not many, missions, the choice for an LEO to GEO stage is reduced to a competition between the nuclear and chemical engines. If the number of

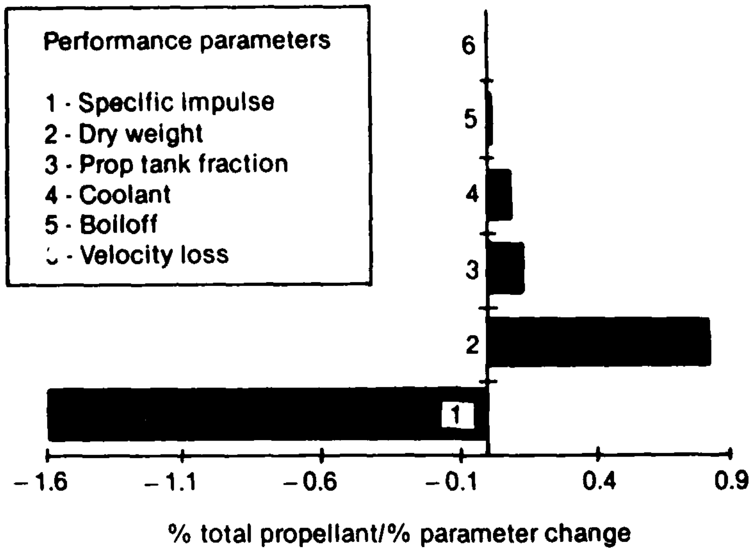

Figure 9. Impact of performance parameters on total mission propellant--LEO-GEO nuclear stage.

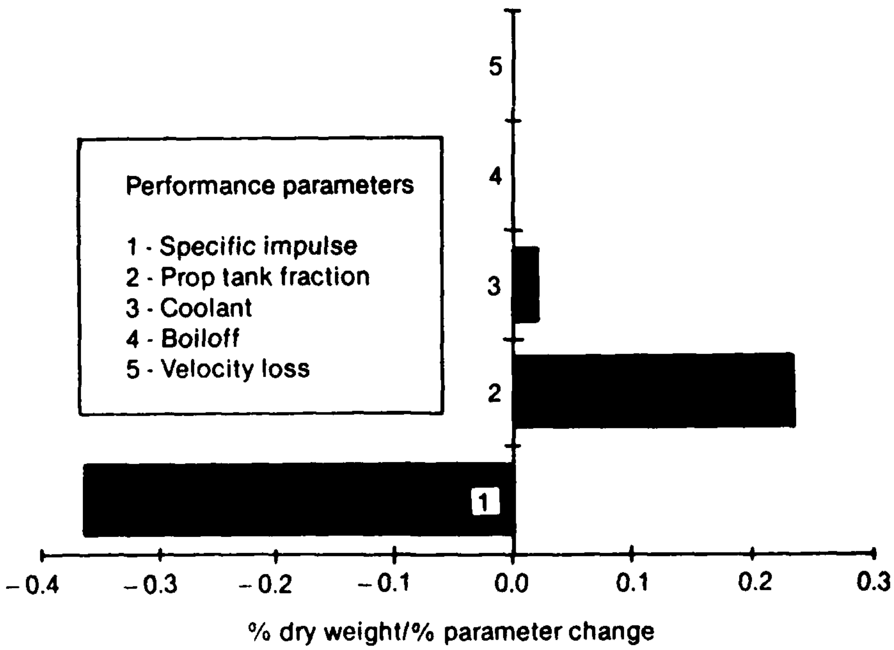

Figure 10. Impact of performance parameters on vehicle dry weight--LEO-GEO nuclear stage.

TABLE 5. LEO-GEO NUCLEAR STAGE OPERATIONS COST SENSITIVITY (Cost basis-20 flights)

Parameter	Baseline	Operations Cost Sensitivity (At baseline)	
		$/Parameter Units	Percent
Payload-to-Orbit Cost, $/lb	750	0.67M	0.91
Reliability, %	99.5	-7.5M	-1.0
Engine Life, Flights	80	-0.1M	-0.01
Structure Life, Flights	100	-0.1M	-0.01
Propellant Scavenged, %	60	-5.0M	-0.68

TABLE 6. LEO-GEO LIFE CYCLE COSTS

Life Cycle - Number of Missions	Life Cycle Costs, $B		
	Nuclear	Chemical	Electric
20	2.72	1.98	2.9
100	4.97	5.36	5.8
500	15.9	22.0	16.9
1000	29.4	42.6	31.3
1542 - Mission Model A[a]	43.8	64.8	46.7
2972 - Mission Model C[a]	81.8	123.2	87.4
3669 - Mission Model B[a]	100.2	151.6	107.0

a. Note--see Section 1 for descriptions; briefly, the bases are
A-STAS Option III/3 (Jan. 86)--no integral stage payloads
B-STAS Option III/3 (Jan. 86)
C-DOD Option 3 (Dec. 86)

missions is small, e.g., less than 75, the chemical engine LCC is less than that of the nuclear engine and, barring any operational demands for which the nuclear engine is uniquely qualified, there is no incentive for undertaking development of a nuclear engine and stage. The situation changes as the number of missions increases and, for the numbers of missions represented by the three mission models, the potential savings indicated for the nuclear stage are very large. The bottom numbers in the table indicate the nuclear stage has a $50B advantage over chemical for the

large number of missions involved in Mission Model B representing unmanned missions in DOD Option 3 (partial SDI deployment--see Section 1).

The preliminary assessment of the relative merit of nuclear, chemical and electric engines for support of Lunar and Mars bases is based on utilization of the same engine characteristics as those for the LEO-GEO vehicles. The required vehicles were defined and the mission propellant requirements determined. Since propellant-in-LEO requirements are the major cost contributors to these missions, they, along with key operational characteristics (e.g., transfer time), form a basis for comparing the relative attractiveness of the competing engines. Table 7 provides the propellant requirements for the more attractive vehicle combinations for

TABLE 7. MISSION/STAGE PERFORMANCE SUMMARY

Mission/Stages	One-Way Transfer Time (days)	Propellant Required (lb)
LEO-GEO		
Nuclear	1	25,394
Chemical	1	53,000
Electric	58	7,347
Lunar Base Support		
Nuclear OTV (w/o aerobrake)/Nuclear LDAV	3	92,600
Chemical OTV (w aerobrake)/Chemical LDAV	3	243,800
Electric OTV (w/o aerobrake/Chemical LDAV	>300	90,000
Mars Base Support		
Nuclear Taxi (w aerobrake)/Nuclear MDAV	200	131,000
Nuclear Taxi (w aerobrake)/Chemical MDAV	200	214,000
Nuclear Taxi (w/o aerobrake)/Nuclear MDAV	200	254,000
Nuclear Taxi (w/o aerobrake)/Chemical MDAV	200	376,000
Chemical Taxi (w aerobrake)/Chemical MDAV	200	558,000
Electric Taxi (w/o aerobrake)/Chemical MDAV	>700	255,000
CASTLE/Cycling Orbits		
Nuclear Taxi (w aerobrake)/Nuclear MDAV	<200	428,000
Nuclear Taxi (w aerobrake)/Chemical MDAV	<200	619,000

the Lunar and Mars missions. (Other combinations are included in the discussion in Sections 2.4 and 2.5.) Trip transfer times are noted because of the significant bearing on selection of viable alternatives. Data for the LEO-GEO mission are also included.

The propellant requirements for the three options for Lunar base support indicate that both the nuclear and electric vehicles would have a large operational cost advantage over the chemical stages because of the latter's large propellant requirements. The long transfer time of the electric OTV provides no incentive to select it, and thus the nuclear OTV/LOAV combination is the preferred choice.

For the Mars base support there are more options to consider, but the nuclear engine is again indicated as the preferred choice. Considering the large propellant requirements of the chemical taxi and the long transfer time of the electric propulsion vehicle, they do not provide reasonable choices over the nuclear engine.

Concluding this brief summary of the mission trade-off studies, it is noted that consideration of the Lunar base and Mars base support provides further support for development of nuclear rocket engines for space transportation. If the Air Force undertakes its development for application to Earth orbital missions, the resulting product should also be attractive for other space missions including support of Lunar and Mars bases.

Safety considerations concerning the use of nuclear rocket engines for space propulsion are reviewed in Section 2.6. The review concludes that a nuclear rocket engine in an orbital transfer vehicle or tug is inherently safer than a chemical engine. Furthermore, U.S. and U.N. policies exist for the use of nuclear reactors in space. Guidelines for space reactors have been prepared by DOE and NASA in more recent years, and a safety review and approval process is now in place. A review of the nuclear rocket development program (Rover/NERVA) that was conducted from the mid-fifties until 1973 leads to the philosophy that flight safety considerations should start with the design process and provide solutions

that are built into the design. The NERVA program resulted in very detailed specifications and criteria for the safe use of rocket reactors. Design practices and an extensive experience base that is still applicable today were developed. This technology base provides important information to be used in the development and use of any future nuclear rocket engine as well as the confidence that nuclear rockets can be safety developed for future space missions.

An in-depth review and analysis of the safety issues involved in the use of the ANRE in Earth orbit transfer missions is provided in Appendix I. The review concludes that the probability of a serious failure in each mission (one startup and one shutdown) is low, i.e., between 0.5 and 1.3 per thousand. For the relatively large numbers of missions involved in the selected mission models, this probability approaches one. Therefore, planning for a safe recovery from all failures is an absolute program requirement and must be an integral part of all mission and design studies.

2.2 Mission Analysis

In support of the Advanced Nuclear Rocket Engine (ANRE) study, different classes of DOD and Civil missions have been reviewed. These missions are identified by the mission models furnished for the Space Transportation Architecture Study (STAS), that is, the DOD Space Transportation Mission Requirements Definition, Issue 7, dated December 12, 1986, and the Civil Needs Data Base (CNDB), Version 2.0, dated January 24, 1987. The classes of DOD and Civil missions reviewed for this study are: DOD Option 2, which identified the normal growth military launch scenario; DOD Option 3, which adds to the normal growth scenario those missions expected for Strategic Defense Initiative (SDI) Kinetic Energy Weapon (KEW) deployment; and DOD Option 2 and Civil Option II, which combines the military normal growth scenario and the civilian nominal growth model.

The missions have been categorized according to payload mass and required delta velocity, in order to identify probable mission requirements

for a nuclear rocket-propulsion orbit transfer system. The results of this effort are summarized in Tables 8, 9, and 10. Only those payload events requiring an upper stage, that is, requiring a delta velocity greater than zero, are included in these tables. The payload event counts included for years 2011 through 2020 are extrapolated from the events for years 2006 through 2010. Of the 668 Civil Option II payload events included in Table 10, 541 have both a delivery mass and a (not necessarily equal) return mass. Manned and return-only payload events have been excluded from this summary. Both retrieval operations, required for the return of payloads and man, and man-rating requirements fall outside the scope of this study.

The destinations for the missions summarized in Tables 8, 9, and 10 may be grouped as shown in Table 11. The mission destinations other than Lunar and planetary are considered to be Earth orbital transfer missions. The Lunar and planetary missions, contained in Civil Option II, are further described in Table 12.

This mission analysis has been used to size a baseline upper stage for Earth orbital transfer missions. The upper stage has been sized to deliver 14,000 lb from Low Earth Orbit (LEO) to geosynchronous orbit (GEO). Such a stage will handle 100% of the payload events summarized in Tables 8 and 9, for DOD Options 2 and 3, respectively. It will handle over 98% of the 668 delivery payload events summarized in Table 10 for Civil Option II. The Civil payload events not handled by such a stage are either GEO payloads that are too heavy or planetary payloads. Within the groundrules adhered to on STAS, the too heavy GEO payloads may be modularized and then transported by this baseline stage. Thus, the baseline stage captures 100% of the Earth orbital payload delivery events from the three mission classes.

The nuclear upper stages are expected to be operated from two space-based platforms, one each for payloads launched from the Eastern Test Range (ETR) and the Western Test Range (WTR), respectively. For STAS, ETR-launched payloads were delivered to park orbits of 28.5 and 57 degrees inclination; WTR-launched payloads were delivered to park orbits of between

TABLE 8. DOD OPTION 2: UNMANNED PAYLOAD EVENTS, 1995-2020

	Payload Raw Mass (lb)				
δ V (fps)	0-7000	7000-14000	14000-21000	21000-28000	28000-35000
0-2800	26	66	33	--	51
2800-5600	27	--	--	--	83
5600-8400	119	--	--	--	--
8400-11200		78	61	--	--
11200-14000	54	96	--	--	--

TABLE 9. DOD OPTION 3: UNMANNED PAYLOAD EVENTS, 1995-2020

	Payload Raw Mass (lb)				
δ V (fps)	0-7000	7000-14000	14000-21000	21000-28000	28000-35000
0-2800	1968	3982	257	--	51
2800-5600	27	--	--	--	83
5600-8400	119	--	--	--	--
8400-11200	--	78	61	--	--
11200-14000	28	137	--	--	--

TABLE 10. CIVIL AND DOD UNMANNED PAYLOAD EVENTS, 1995-2020

	Payload Raw Mass (lb)				
δ V (fps)	0-7000	7000-14000	14000-21000	21000-28000	28000-35000
0-2800	582	78	42	--	52
2800-5600	27	--	--	--	83
5600-8400	119	--	--	--	--
8400-11200	4	78	61	--	--
11200-14000	112	113	2	3	--
14000-16800	1	4	--	--	--
16800-19600	1	--	--	--	--

TABLE 11. MISSION DESTINATIONS

Mission Destination	Unmanned Payload Events, 1995-2020 DOD2	DOD3	DOD2/Civil II
Space Station Orbit Vicinity (28.5°)	0	0	53
Low Earth Polar and Sun-Synchronous (90-99°)	113	863	638
Mid-Inclination, Altitude Below 1000 nm (50-80°)	192	5524	192
Mid-Inclination, Altitude Above 1000 nm (50-80°)	265	306	265
Geosynchronous and Near-Geosynchronous (0-10°)	124	98	201
Lunar and Planetary	0	0	13

TABLE 12. CIVIL OPTION II LUNAR AND PLANETARY MISSIONS

Lunar and Planetary Missions	Events	Payload lb/Event
Lunar Relay (Lunar Backside Communications)	4	2,500
Venus Atmosphere Probe	1	1,760
Mars Sample Return	2	20,790
Planetary (Other) - European Space Agency	1	2,500
Saturn Orbiter/Titan Probe	1	8,800
Comet Sample Return Mission	3	7,106
Saturn Fly-by/Probe	1	3,300

70 and 99 degrees inclination. For the DOD Option 2 payloads, summarized in Table 8, 54% are to be launched from ETR and 46% are to be launched from WTR. For the DOD Option 3 payloads, summarized in Table 9, 5.7% are to be launched from ETR and 94.3% are to be launched from WTR. For the DOD Option 2 and Civil Option II payloads, summarized in Table 10, 38% are to be launched from ETR and 62% are to be launched from WTR.

Optimal placement of space-based platforms for the nuclear upper stage is dependent on the specification of groundrules for manifesting payloads on the upper stage. These groundrules need to identify such details as: the weight of the propellant required for delivery of the payloads and for return of the upper stage to its platform; the maximum propellant weight allowable for the upper stage; the additional weight required for payload attachment structure; the maximum number of payloads that can be delivered by a single upper stage; and, how closely spaced the destinations must be for payloads delivered by the same upper stage. Such details are not available within the scope of this report.

The Earth orbital transfer missions have been analyzed to determine the amount of inclination change involved in the transfer from park to operational orbit. In this analysis, we have used the park inclinations employed by STAS. The GEO and near-GEO payload events require an inclination change of up to 28.5 degrees. Except for 90 of the DOD Option 2 and DOD Option 3 non-GEO mid-inclination higher-altitude payload events, which require an inclination change similar to that for GEO, the remaining Earth orbital missions require a change of 8 degrees or less. Thus, there is not a substantial need for an upper stage specifically designed for large inclination changes.

The near Earth (non-GEO) missions have also been analyzed to assess the need for a second baseline stage design. A stage, sized for a 7000 lb payload and a delta velocity of 8400 ft/sec, could deliver a substantial number of the near Earth payloads with less total propellant than required by the stage sized to deliver 14,000 lb to GEO. The payloads deliverable by this smaller stage are those weighing 21,000 lb or less and requiring appropriate delta velocities. The exact propellant savings obtainable by using the two differently sized nuclear upper stages is dependent on the specification of groundrules for manifesting payloads on the two upper stages. Such groundrules are not available within the scope of this study.

2.3 Earth Orbital Transfer Missions

The LCC analysis reported in Section 1 are updated here to the mission models described in that report. Those analyses are also extended to provide a comparison of the ANRE to NEP in addition to the updated comparison to chemical propulsion. This chapter also provides the results of parametric analyses for the LEO to GEO ANRE stage design to show the sensitivity of key design and operational parameters on dry weight, propellant weight, operational costs, and life cycle costs.

2.3.1 Typical Orbital Transfer Missions

Figure 11 is a schematic of the various stages of a typical Earth orbit transfer and the major elements involved. The elements consist of a tug (Orbit Transfer Vehicle-OTV), service platform (a maintenance facility in a circular orbit around the Earth), robot (chemically powered element that is normally attached to the tug for precise maneuvering and certain types of maintenance, spacecraft (payload, such as a satellite, that is to be positioned), central control (control station that has overall responsibility for the transfer), and orbital traffic control center (a backup control station that has real-time data concerning the total space environment). The robot referred to above is optional; the tug could also be maneuvered using only its attitude control devices (reaction control system).

It is assumed that most payloads from the earth are delivered to the platform by the Shuttle or Large Cargo Vehicle (LCV). The tug (OTV) can be assigned a number of tasks; it can deliver single payloads or multiple payload pallets to higher Earth orbits, geosynchronous Earth orbit (GEO), or multiple-GEO destinations. It is also able to retrieve payloads from higher orbits, delivering them to the service platform for maintenance and repair or to the Shuttle for return to Earth.

Initially (see Figure 11, Part A) the tug is assumed to be docked at the service platform. After the spacecraft is launched from Earth, either the robot detaches and docks the spacecraft or the Shuttle (on LCV) docks

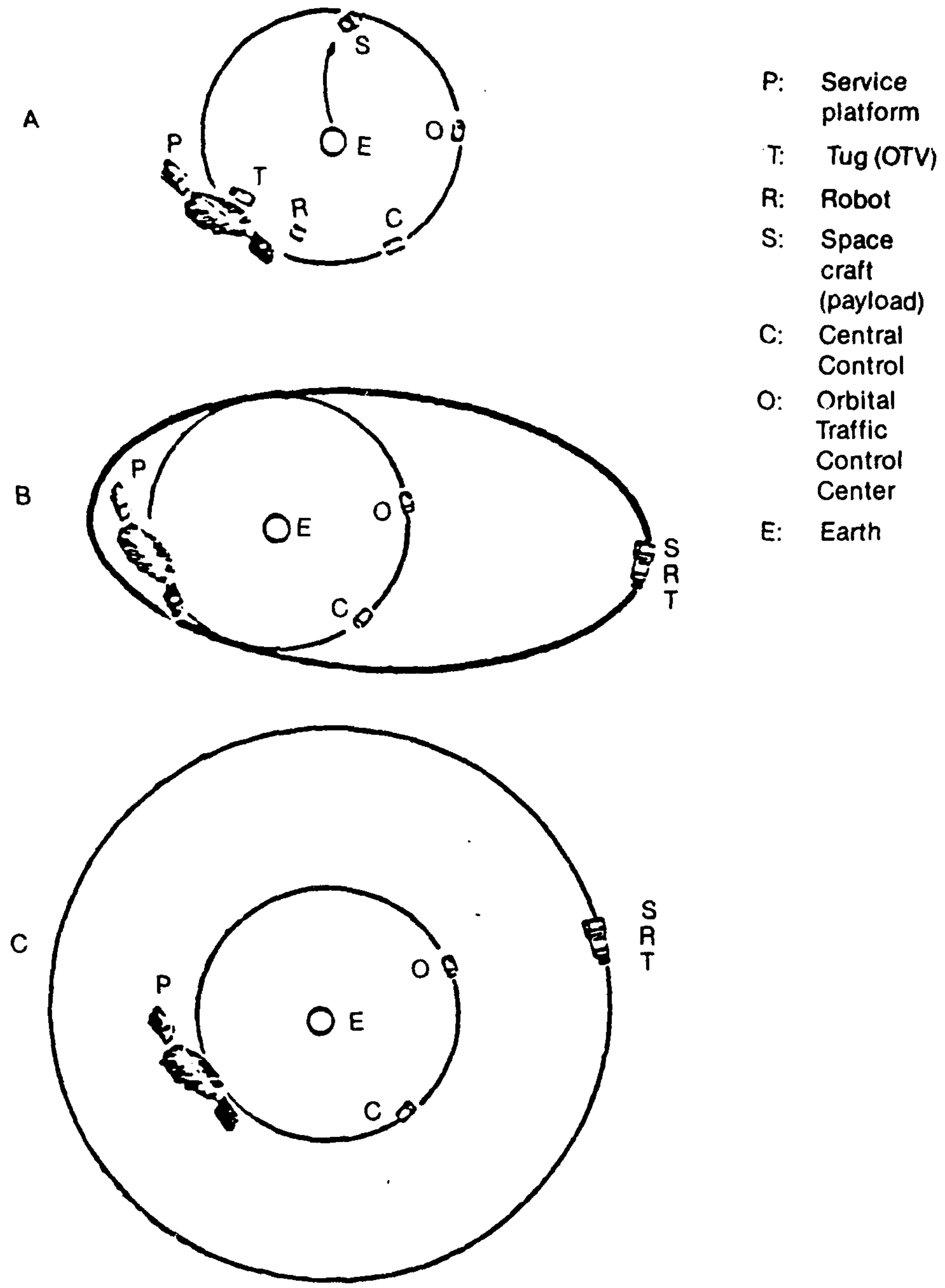

Figure 11. Earth orbital transfer mission.

at the service platform. After any work or checkout is completed on the spacecraft at the service platform, the spacecraft is ready for transfer to a higher orbit. The robot maneuvers the tug and spacecraft away from the service platform to a distance that is safe for initiating the first burn of the tug, and the combination is placed in an elliptical orbit (Figure 11, Part B). At the appropriate time, a second burn is initiated in order to attain the desired circular orbit (Figure 11, Part C). Finally, the robot is used to detach the tug from the spacecraft and move the tug to a safe distance. The process is then reversed and the tug is returned to the service platform.

If the mission is to retrieve the spacecraft from another orbit and return it to the service platform, a similar procedure would be followed. The robot would maneuver the tug to a safe distance from the platform. The tug would then execute the two burns to arrive at an orbit near the spacecraft. The tug would shut down; the robot would deploy, retrieve the spacecraft, and reattach to the tug. Upon returning to a safe distance from the platform, the tug shuts down. After receipt of clearance, the robot maneuvers the tug and spacecraft for docking at the platform.

2.3.2 Baseline Nuclear Stage Design

Two nuclear stage designs were generated on the basis of the two major mission categories described below. The LEO to GEO and Return mission stage design is considered the baseline configuration. The Near Earth mission stage design was conducted in order to assess the propellant penalty associated with using the larger stage for the Near Earth missions.

- LEO to GEO and Return: This mission requires the transportation of 14000 lb of payload from a LEO (assumed as a 150 nautical mile circular orbit inclined at 28.5 degrees) to a GEO and returning the transfer vehicle back to the LEO. Durations for the Hohmann transfer orbit and stay times at destinations were assumed to be the same as a comparable cryogenic OTV. This stage can also be used to conduct the near Earth missions at a penalty of off-loading propellant.

- Near Earth Missions: This mission category established a requirement for the delivery of 7000 lb of propellant using 8400 ft/sec in delta velocity, then returning the vehicle to origination point by using another 8400 ft/sec in delta velocity. This particular mission capability allowed capture of all the low delta velocity missions in the mission model.

General Sizing Approach. The sizing procedure for the nuclear stage design was largely the same as that used for Task I activities. This procedure utilizes the ideal rocket equation and estimates of various subsystem weights to determine the total propellant required to accomplish the mission.

The procedure was automated for Task II in order to allow rapid evaluation of performance sensitivities. The sizing model now incorporates further detail. Estimation algorithms for propellant boiloff ullage and secondary propellant usage (for subsystems such as fuel cells) were generated on the basis of previous cryogenic OTV work; techniques for handling mission energy losses were also added. Input data for the sizing model now includes: mission delta velocities, mission payload(s), mission phase durations, subsystems weight estimates, propellant boiloff characteristics, selected propellant usage rates for fuel cells, auxiliary propulsion and engine cooling, and engine performance characteristics.

The actual sizing procedure for each baseline system relies upon the specific input data and the assumptions used for optimization, operational characteristics, etc. Sizing proceeded by using the input conditions to establish the desired configuration characteristics and mission requirements. Iteration, using the ideal rocket equation, proceeded over all phases of the mission until estimated propellant requirements were sufficient to satisfy the required performance to the desired level of sensitivity.

Because of the introduction of the propellant boiloff algorithm, it is not sufficient to iterate the model until mission performance is satisfied. It is necessary to modify the thickness of the Multi-Layer

Insulation (MLI), and thus the propellant boiloff rate, in order to determine, to first approximation, the optimum configuration. Different vehicles are sized to the same basic requirements for different thicknesses of MLI. That thickness which results in the lowest total propellant weight, the highest cost item in our operational scenario, is selected as optimum.

Groundrules and Assumptions. The baseline performance for the NERVA derivative ANRE is noted in Table 13. The nature of the revision to the NERVA reactor is described in the Task 1 report; it primarily consists of major reductions in engine size and thrust, as well as use of advanced materials to reduce engine weight and use of advanced fuel elements to improve specific impulse.

TABLE 13. BASELINE ADVANCED NUCLEAR ROCKET ENGINE (ANRE) DATA

Engine Dry Weight (lb)	4,600
Chamber Pressure (psi)	3,070
Thrust (lb force)	14,550
Flow Rate (lb/sec)	15
Engine Length (ft)	8.9

Mission analysis provides the payload and delta velocity requirements for the baseline stage designs.

Vehicle systems were not length limited and maximum diameter was set at 14.2 ft, consistent with projected launch vehicle payload container limits.

A constant value of 20% contingency of total dry weight was used in sizing all configurations. This contingency value is consistent with advanced launch vehicle design and represents margin for sizing inaccuracies and uncertainties in specific advanced technologies.

A flight performance reserve, representing a percentage of the total delta velocity, of 2% was required for all configurations.

Liquid hydrogen was the assumed fuel for all configurations.

The effects of the shutdown cooling propellant, propellant used to cool the reactor after the main burn, on vehicle performance was estimated by applying the same procedure used for Task I; reported engine specific impulse was adjusted for impulse contributions by cooling propellant while the cooling propellant itself was calculated based upon main burn time and considered ejected after the burn.

The pulsed propellant flow, which is closed loop, used in the later phases of cooling was presumed to remain onboard the vehicle to charge accumulator tanks for the auxiliary propulsion system.

Each stage design was optimized for MLI on the basis of total propellant, which includes the tank boiloff propellant.

In addition to the general assumptions concerning stage geometry, both stages utilize a square root of two elliptical dome construct for both ends of the main tank.

<u>LEO to GEO Stage</u>. The basic requirements for this stage are to deliver 14,000 lb of payload from LEO to GEO orbit and then return the vehicle, without payload, to LEO. In addition to these requirements, durations associated with the different phases of the mission, such as prelaunch phase, first burn, first coast, were determined from orbital mechanics and assumptions previously used in the cryogenic OTV design work. The durations and mission phases are defined in Table 14. Finally, estimates of the velocity losses for this mission were taken from a previous chemical OTV design analysis. Accurate estimates for this parameter were not necessary since it has little impact on the overall stage design for this mission; see the parametric study results for more information.

Subsystem Assumptions - To generate the input requirements for the sizing model, the following assumptions were made concerning the subsystems of the vehicle:

TABLE 14. LEO-GEO MISSION PHASES

Phase Number	Description	Duration (hours)
1	Wait for mission	0
2	Burn 1 and coast	12
3	Burn 2 and position	6
4	Offload and wait for return	16
5	Burn 1 and coast	12
6	Burn 2 and position	3
7	Wait for pickup	6

- The propellant tankage fraction, ratio of tankage weight to contained propellant, was set at 0.075, which includes the 20% contingency factor.

- Multiple-Layer Insulation (MLI) was assumed to provide insulation of the main tank to limit boiloff of the fuel. Actual thickness and weight of the MLI were based upon optimization of propellant weight.

- Propellant boiloff characteristics were taken from liquid hydrogen tank data for cryogenic OTV design.

- Usage rates of propellant for fuel cell (oxygen provided by separate tankage) were estimated for similar cryogenic OTV designs to be 0.7 lb/hr.

- Propellant used for attitude control was estimated as 2% of the main propellant used during the specific mission phase.

- Ullage was estimated at 1.5% of the total volume while residual propellant was estimated as 1.5% of the total main propellant required. Total tankage volume satisfies total propellant, main plus residuals, as well as ullage.

The weight assumptions for specific vehicle subsystems are listed in Table 15.

TABLE 15. SUBSYSTEM WEIGHTS FOR LEO TO GEO STAGE (lb)

Avionics	400
Shielding	100
Environmental Control	150
Structure	411
Auxiliary Propulsion and Fuel Cell	350

Near Earth Stage. Although the baseline configuration, the LEO to GEO stage, is capable of handling the near Earth missions, it does so at a penalty. If single payload delivery only is considered, propellant must be off-loaded from the heavy stage. The stage then must drag along the extra unused tankage and this requires additional propellant that a specific point design stage would not have to use. The exact penalty is a function to a specific mission. In order to assess the specific propellant penalties, a specific stage design for the near Earth missions was developed to compare to the off-loaded LEO to GEO stage.

Only three mission phases were assumed to simplify the analysis. The duration time of the prelaunch phase was assumed to be 24 hr while the other two phases were assumed to be 5 hr each. It is assumed that the payload is dropped off at the destination while the stage returns to the initial point.

Subsystem Assumptions - To generate the input requirements for the sizing model, the following assumptions were made concerning the subsystems of this vehicle:

o MLI was assumed to provide insulation of the main tank while boiloff rates were taken from liquid hydrogen tank design for a chemical OTV. (Optimization for MLI thickness proceeded in the same manner as for the LEO to GEO stage. Due to the shorter

mission duration period, the MLI thickness optimized to 0.05 inches, an order of magnitude lower than for the larger stage.)

- Rather than fuel cells, used in the baseline configuration, this design was assumed to use batteries that are charged by the supporting platform between missions.

- Propellant used for attitude control was estimated as 2.5% of the main propellant used during the specific mission phase. The greater amount, relative to the baseline configuration, was deemed necessary for proper payload positioning requirements.

- Ullage and residual requirements were assumed to be slightly greater than for the LEO to GEO stage.

- The propellant tankage fraction was assumed to be the same as for the LEO to GEO stage.

The weight assumptions for specific vehicle subsystems are listed in Table 16.

TABLE 16. SUBSYSTEM WEIGHTS FOR NEAR EARTH STAGE (lb)

Avionics	350
Shielding	50
Environmental Control	150
Structure	411
Auxiliary Propulsion and Batteries	400

2.3.3 Comparison of LEO-GEO and Near Earth Nuclear Stages

The weight statements for the LEO-GEO and Near Earth stages are presented in Table 17. Figure 12 shows the configuration of the LEO-GEO (baseline) stage. The configuration of the Near Earth stage is similar except for its length (and use of batteries in lieu of fuel cells).

TABLE 17. NUCLEAR STAGE WEIGHT STATEMENTS

	LEO-GEO Stage (Baseline)	Near Earth Stage
Gross Flight Weight, lb	48,779	24,129
Total Propellant, lb	25,394	9,269
Shutdown Coolant, lb	2,000	500
Boiloff, lb	185	36
Dry Weight, lb	9,385	7,860
Tankage Weight, lb	1,905	695
Stage Length, ft	48.54	25.5

As previously mentioned, the use of the LEO to GEO stage for near Earth missions is feasible but exacts a penalty. In order to assess this penalty, the propellant required by the LEO to GEO stage to satisfy the near Earth mission requirements was calculated. The difference between this value and the propellant necessary for the near Earth stage is the single payload penalty for use of the LEO to GEO stage. Conservatively, this penalty could be multiplied by the number of missions and the delivery cost of propellants in order to ascertain the cost penalty associated with using the LEO to GEO stage for less than optimum missions. This estimate is affected by the number of other missions using even less delta velocity and the number of payloads that could possibly be carried. Obviously, limiting the LEO to GEO stage to a single payload is impractical for near Earth missions, but the maximum number is limited by structural considerations, traffic requirements, and payload destinations.

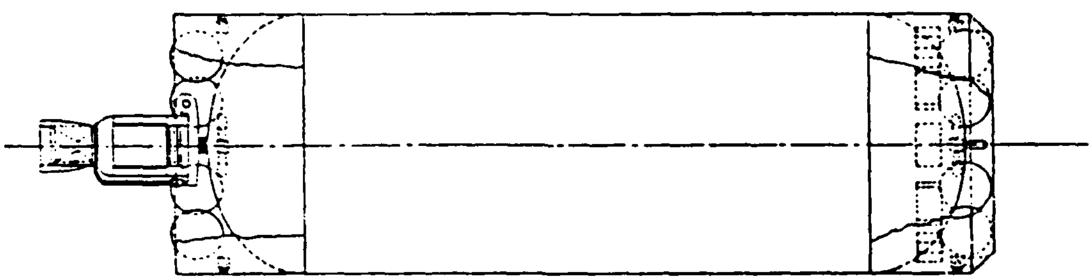

Figure 12. Baseline LEO-GEO nuclear stage configuration.

The exact propellant penalty was 957 lb per single payload mission; the mission being 7000 lb of payload delivered via 8400 ft/sec up and back. This single payload mission penalty multiplied by the total number of these missions in the current mission model over the total duration of 26 years exceeds 100,000 lb. Taking into account the lower delta velocity missions that must also be satisfied, the total propellant penalty alone would exceed $100M (at current launch cost) spread over 26 years. This value would be magnified if small delta velocity missions had to be accomplished by the LEO to GEO stage. Thus, it seems appropriate to have a second (smaller) nuclear propulsion stage for the near Earth missions.

2.3.4 Parametric Analysis for Nuclear LEO-GEO Stage

The design parameters selected for the LEO to GEO stage parametric analysis include: specific impulse, cooling propellant lost, dry weight increase, change in velocity loss for the mission, boiloff rate, and the propellant tankage fraction. These parameters were selected on the basis of their impact on the sizing analysis, as characterized by total propellant and dry weight, and performance.

The procedure used for conducting parametric analysis for the various configurations is similar to that for launch vehicle analysis. Basically, for a parameter of interest, the sizing model is run after varying the parameter slightly and keeping all other parameters fixed. The resulting configuration is a slight variation of the baseline. Resulting data are used to establish the sensitivity of the baseline to the parameter of interest. In each case the selected parameters were varied and the resulting configuration data were plotted versus each parameter. The resulting plots are nearly linear, particularly near the baseline value and thus the numerical results are provided in Table 18. The graphical results are presented in Appendix F.

For the LEO to GEO configuration, there are three parameters that cause a significant variation in propellant weight, the major cost item. These are the specific impulse of the stage, the dry weight, and the

propellant tankage fraction. The configuration is largely insensitive to all the other parameters.

Table 18 indicates the configuration sensitivity of propellant weight and dry weight to specific impulse and the propellant tankage fraction. Although configuration dry weight sensitivity is of less interest, due to the predominance of propellant in cost estimates, it is presented as well. The configuration has roughly a 42 lb/sec propellant sensitivity to specific impulse. This is a large value. Note, however, that the dry weight is much less sensitive to specific impulse. On a percentage basis, the sensitivity of propellant weight to dry weight is second in importance after specific impulse while propellant tank fraction is third. Due to the nature of the tankage fraction, which is multiplied times the total propellant to determine tankage weight, the configuration is more sensitive to this parameter than to the remaining variables. The shutdown cooling, which is unique to nuclear rocket engines, is the only one of the remaining

TABLE 18. PARAMETRIC ANALYSIS SENSITIVITY RESULTS

	lb/parameter units	%/%
Total Mission Propellant Weight		
Parameter		
Specific Impulse (sec)	-41.7	-1.591
Dry Weight (lb)	2.117	0.805
Propellant Tank Fraction (%)	461.3	0.135
Shutdown Cooling Propellant (lb)	2.215	0.087
Boiloff Propellant (lb)	2.207	0.016
Velocity Loss (FPS)	2.920	0.010
Vehicle Dry Weight		
Parameter		
Specific Impulse (sec)	-3.51	-0.363
Propellant Tank Fraction (%)	292.9	0.234
Shutdown Cooling Propellant (lb)	0.187	0.020
Boiloff Propellant (lb)	0.186	0.004
Velocity Loss (FPS)	0.246	0.002

parameters analyzed that has a significant influence on propellant requirements. The effects of propellant boiloff and vehicle velocity loss are insignificant.

As a final note, the sensitivity of the stage to engine thrust was not investigated. For the selected mission category, the baseline engine has sufficient thrust to accomplish the mission. Due to nuclear reactor size limits, reduction of the thrust by reducing the reactor size is not easily accomplished, conversely higher thrust levels merely increase engine dry weight while not adding any additional performance. The velocity loss terms, which are reduced at higher thrust levels, are not large enough to have any significant effect on the stage.

2.3.5 Nuclear Stage Operational Cost Sensitivities

Operational cost sensitivities were developed for the nuclear stage for various design parameters. They include delivery cost to orbit, reliability, structure and engine life, and propellant scavenging. Each of these sensitivities is discussed in the following paragraphs. The operations costs are based on 20 missions, but the sensitivities would not change significantly for more or less missions.

Delivery Cost to Orbit Sensitivity. Figure 13 shows the impact of varying the cost to deliver hardware and payload to orbit. The cost of delivery to orbit is such a large part of the overall cost that this graph ends up being very close to linear in spite of the learning curve effects on the stage operations and the space based operations. The slope of the line shows that the cost of operations is fairly sensitive to the launch cost. Each change of \$100/lb delivered to orbit represents a change of about \$67M in the operations.

Reliability Cost Sensitivity. Figure 14 shows the effect on operational costs of varying the mission success reliability. This curve basically shows the cost of providing reflight when a mission is aborted. A change

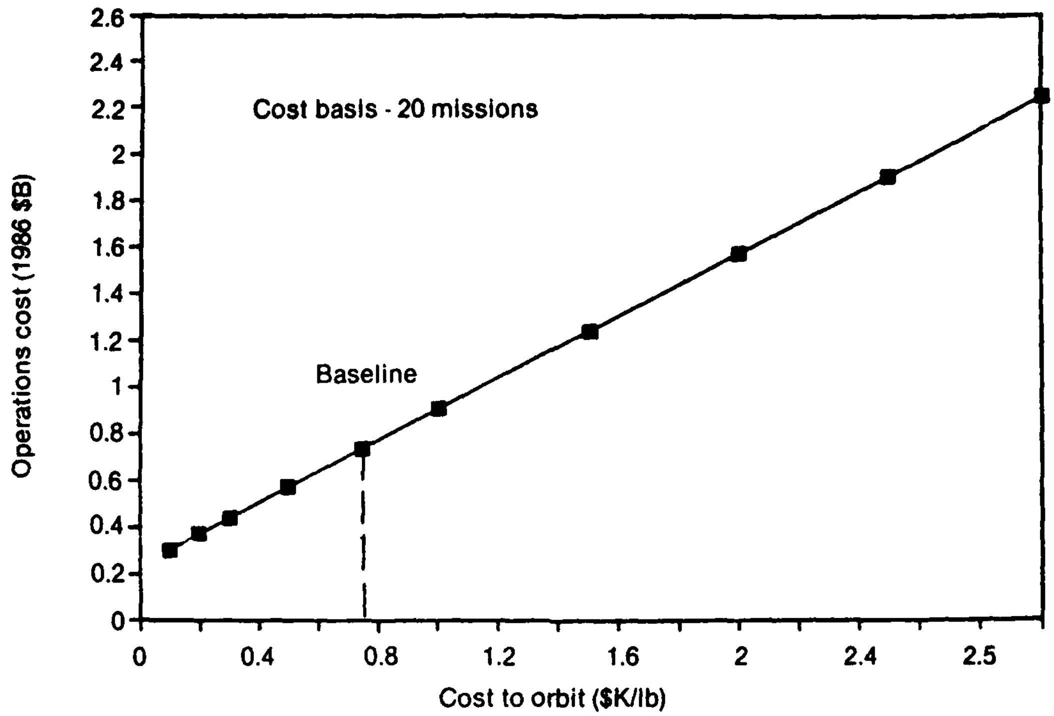

Figure 13. Nuclear LEO-GEO delivery-cost-to-orbit sensitivity.

of one one-hundredth in the mission success reliability represents a change of 7.5M in the operations cost.

Engine Life Sensitivity. Figure 15 shows the operational cost sensitivity of varying the engine life. The optimum life appears to be between 60 and 90 flights. This is where the cost curve begins to flatten and only a slight gain would be realized for the added cost to develop a longer life engine.

Structure Life Sensitivity. Figure 16 shows the operational cost sensitivity of varying the structure life. The optimum life appears to be between 80 and 120 flights. This is where the cost curve begins to flatten.

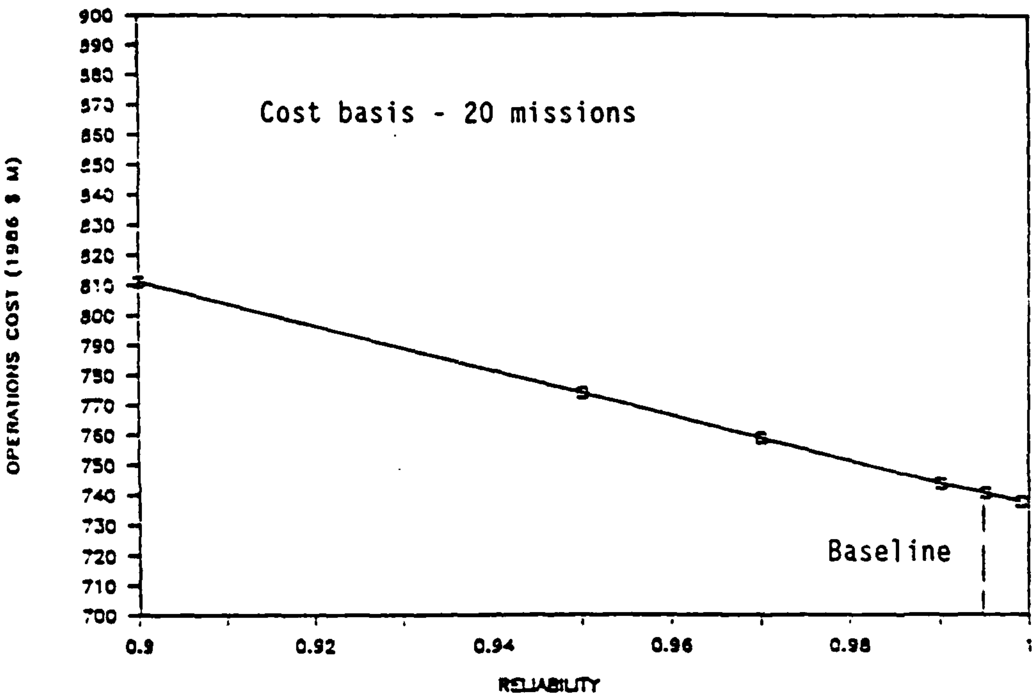

Figure 14. Nuclear LEO-GEO reliability sensitivity.

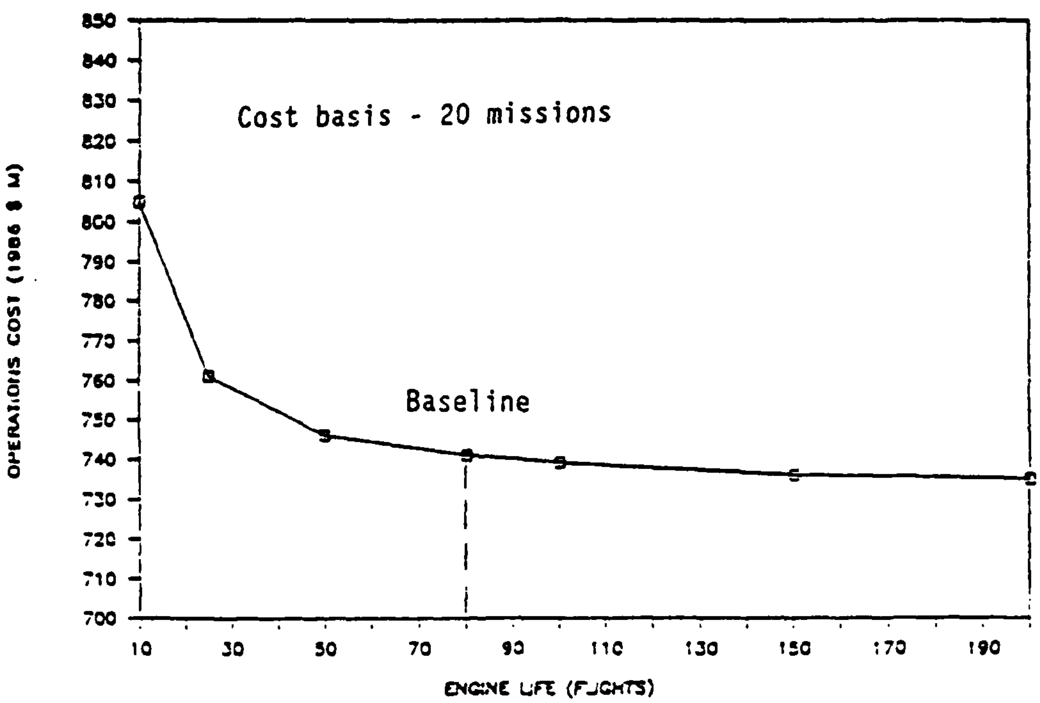

Figure 15. Nuclear LEO-GEO engine life sensitivity.

Propellant Scavenging Sensitivity. Figure 17 shows the cost variation due to scavenging residual propellant from orbital hardware such as the external tank and decaying satellites. The curve shows that for every 10% increase in the amount of propellant scavenged there is a cost savings of $50M. Since the cost of getting propellant to orbit is by far the most expensive single item in the operations cost, any method of reducing the amount that must be transferred from Earth to orbit should be considered.

2.3.6 Life Cycle Cost Analysis

The LCC cycle cost analysis was conducted on a parametric basis to compare the use of a nuclear powered stage against a cryogenic stage and an electrical stage. The costs were broken into three basic phases: design, development, test, and evaluation (DDT&E); production; and operations and support (O&S). The detailed cost breakdowns are provided in Appendix G. The DDT&E costs for the electrical stage are the highest at $2.1B. About half of this cost is for the development of the power system and thrusters. The nuclear stage is the second most expensive at $1.9B. Again, the engine development is about half of this cost. The cryogenic stage costs just under $1B to develop. The cost spread for the cryogenic stage is more in line with the upper stages that are designed today.

The production costs are based on producing two vehicles of each type. More vehicles may be needed depending on specific manifests and flight rates. As more vehicles are built, the average unit cost will drop because of the learning curve effects of large production lots. The electrical stages are the most expensive to build at $198M for the first two units, followed by the nuclear stages at $161M, and the cryogenic stages at $102M. The main engines are the driving subsystem in the nuclear and the electrical stages.

The operational costs for the three vehicles are the baselined at 20 LEO-GEO-LEO flights and a delivery cost of $750/lb to LEO. The stages are assumed to have equal mission success reliability and all benefit from the scavenging of residual propellants. The 20 flights provide a reasonable number of flights over which to assess operational costs. The

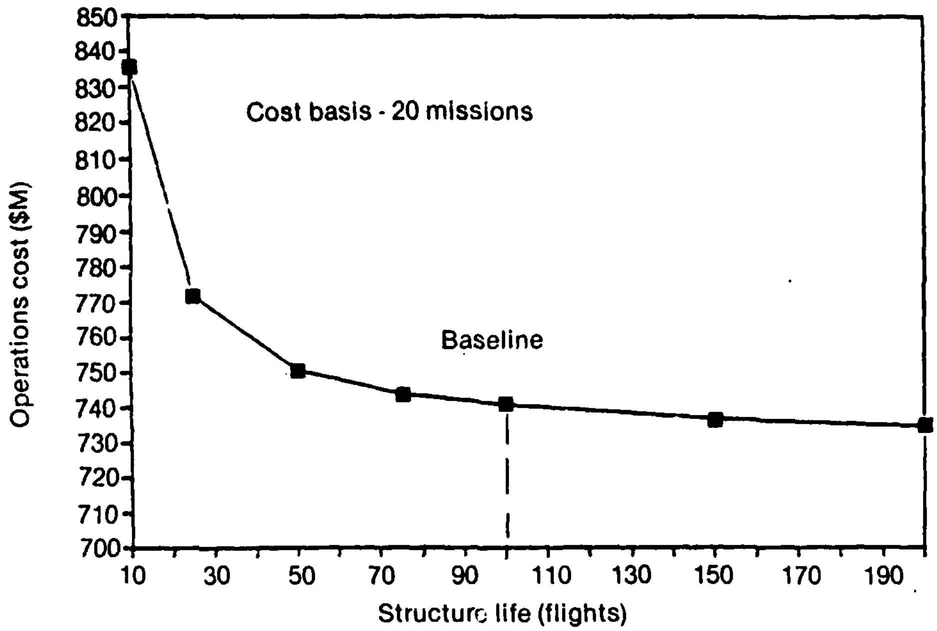

Figure 16. Nuclear LEO-GEO structure life sensitivity.

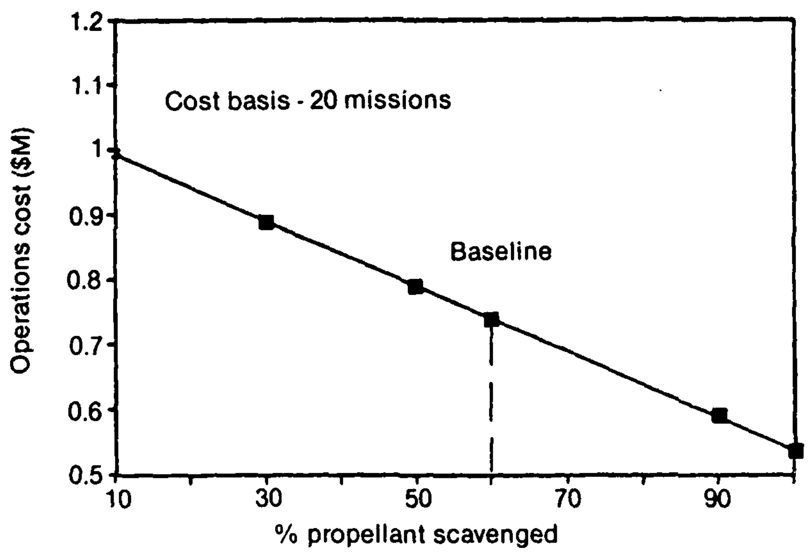

Figure 17. Nuclear LEO-GEO propellant scavenging sensitivity.

20-flight costs for the nuclear, chemical, and NEP stages are 596, 880, and $577M, respectively. The electrical stage is the least expensive to operate mainly because of the tremendous propellant savings, using only 7300 lb of propellant per LEO-GEO-LEO flight compared to 25 Klb for the nuclear stage and 53 Klb for the cryogenic stage. Next following is the nuclear stage. This cost saving of $300M compared to the chemical stage results primarily from the 28 Klb of propellant that is saved over a cryogenic stage.

The LCC was extended out to the the total equivalent missions that are representative of the mission models A, B, and C described in Section 1. The results of this analysis are presented in Figure 18. As would be expected, because of its much lower development costs, the chemical stage is superior if there are not a lot of missions involved in the life cycle. When the number of missions in the life cycle reaches approximately 75, the chemical LCC cost passes the nuclear LCC at a total cost of about $4 billion.

As the number of missions increases, the cost advantage of the nuclear stage over the chemical stage increases quite rapidly, because of the lower propellant requirements of the nuclear stage. The mission models A, B, and C represent approximately 1540, 3670, and 2970 total missions, respectively. The corresponding cost savings of the nuclear stage over the chemical are approximately 21, 51, and $41B, respectively. These cost savings are considerably greater than those estimated in Section 1--the differences are due primarily to the improved cost analysis.

With its somewhat higher development cost, the NEP stage is a little more costly than the ANRE stage initially, and in spite of its much lower propellant requirements, the NEP stage remains more costly as the number of missions increases. For the 3670 missions representing the mission model B, the ANRE saving over the NEP is about $7B out of about $100B total LCC. The much lower propellant needs of the NEP stage compared to the ANRE stage do not make up for the relatively high refurbishment costs determined for the NEP stage because of its long flight time. These are primarily associated with an engine life of 5 missions compared to 80 missions for

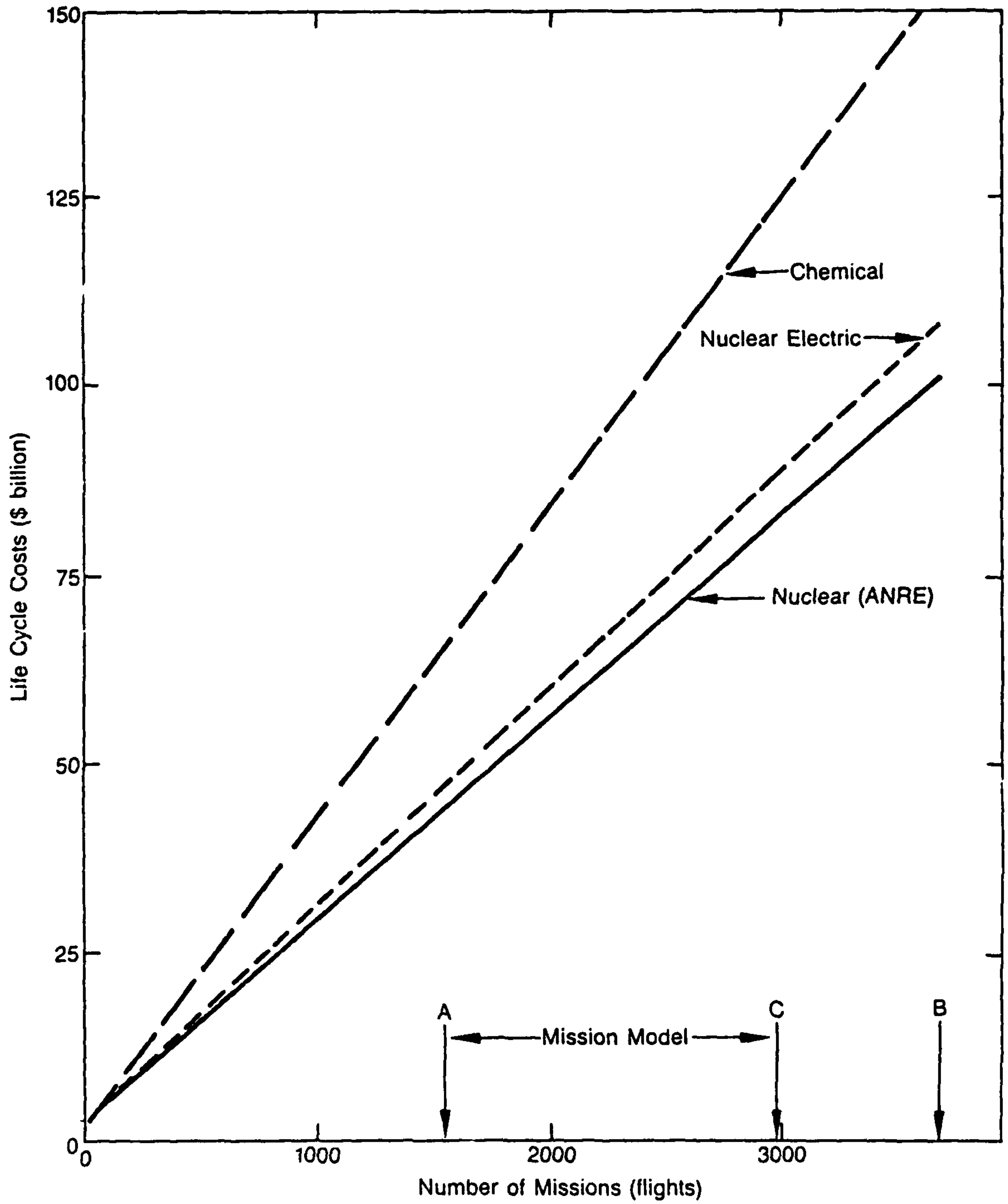

Figure 18. Nuclear LEO-GEO life cycle costs vs. number of missions.

the nuclear stage and the number of spares required for the long mission. Note that for the LEO-GEO-LEO mission the nuclear engine operates at power for less than 30 min, while the NEP engine is at power for many days.

The costing for the LCC study was done at a fairly high level considering the preliminary nature of these studies. Historical program costs were used to develop a baseline methodology for comparing the nuclear propulsion to chemical propulsion systems in the 1995-2020 time frame. Listed below are the groundrules and assumptions that were used to develop estimates for the nuclear OTV and the chemical OTV and OMV.

Groundrules and Assumptions

- All costs are reported in millions of constant 1986 dollars and are exclusive of fees and contingencies.

- During the operational phase, a minimum of two operational stages will be on the platforms at all times.

- The baseline nuclear engine will utilize work from the NERVA program.

- No learning was applied to stage costs due to the small production run.

- The STAS III/3 mission model was used for mission analysis.

- A cost per LCV flight of $70M was used in determining the operations cost.

- Performance capabilities of the LCV are 150,000 lb to LEO and 109,000 lb to Space Station with a payload envelope of 25 ft in diameter and 90 ft in length.

- Mission operations costs were based on a fixed 35 man-year per year level of effort.

- Payload transportation costs were assessed according to the STS reimbursement guide.

- A $250K platform user charge per payload was applied.

- Inter Vehicular Activity (IVA) was charged at $18K per hour per crewman.

- LCV launch costs include the delivery of the initial stages, platforms, and spares.

- Propellant delivery costs were assumed to be 67% provided by hitchhiking and 33% by dedicated tanker.

- Refurbishment costs were spread equally over all missions.

- Service life of the ANRE engine is 80 missions, replacement of other items (structures, avionics, etc.) is assumed at 100 missions.

2.3.7 Comparison of Competing Propulsion Systems

This section summarizes briefly some of the key points that concern the relative merits of the nuclear, chemical, and electric propulsion systems for the conduct of orbital transfer missions summarized in Section 2.2 - Mission Analysis. The significant potential trade-offs among the three propulsion systems are covered in more detail in preceding discussions in this chapter and those in Appendix E - Electric Propulsion Systems. Key data for the nuclear, chemical, and electric stages for the conduct of orbital transfer operations between LEO and GEO are summarized in Table 19. The LEO-GEO mission was selected as a reasonable basis for these early assessments of the relative merits of the compacting propulsion systems in conducting earth orbital missions. All stages have been sized to deliver payloads of 14,000 lb from a service platform in LEO to GEO and return to LEO. The specific impulses of the nuclear and chemical engines are the same as those adopted in the earlier studies reported in Section 1,

970 sec and 475 sec, respectively. The electric engine specific impulse of 4000 sec is based on the review and analysis of NEP summarized in Appendix E.

The data provided in Table 19 and supported elsewhere in this chapter show that if the future earth orbital mission requirements are consistent with those outlined in Chapter 2.2 - Mission Analysis, and the mission models described in the Task 1 report, then the ANRE nuclear engine is a clear choice over both chemical propulsion and NEP to provide the propulsion for these missions. The nuclear engine remains the choice even if the number of required earth orbital missions turns out to be substantially less than indicated by the DOD Option 2, DOD Option 3, and Civil Option II. As indicated by the LCCs in Table 17, the nuclear choice is voided only if the earth orbital mission requirements turn out to be a small fraction of those recently forecast. The chemical engine replaces the nuclear engine as the propulsion system of choice for earth orbital missions only if the number of required missions turn out to be less than 5% of those indicated by Mission Model A.

The NEP engine is not a viable alternate to the ANRE engine for earth orbital missions. As indicated by the LCCs in Table 19, the NEP engine is more costly than the ANRE engine even for large numbers of missions. However, the primary negative aspect of the NEP engine is its long transfer time, about 116 days for a LEO-GEO round trip vs. about 2 days for either a chemical or nuclear engine. The long transfer time of the NEP stage obviates its application to many of the anticipated earth orbital missions.

2.4 Lunar Base Support Missions

Although Section 2.2 - Mission Analysis limited its scope to the analysis of unmanned missions, there is always the need to consider the establishment of a manned Lunar base which would involve the delivery of substantial numbers of relatively large manned and unmanned payloads. This is an application that could readily be satisfied by a nuclear rocket propulsion engine. Accordingly, this section provides a comparative analysis of nuclear, chemical, and electric (NEP) propulsion systems to

TABLE 19. KEY DATA FOR LEO-GEO STAGES

	Nuclear	Chemical	Electric
Payload, lb	14000	14000	14000
Specific impulse, sec	970	475	4000
Thrust, lb	14550	15000	1.5
Power	300 MWt	--	500 kWe
Specific power, kg/kWe	--	--	10
Thruster specific mass, kg/kWe	--	--	5
Round trip transfer time, days	2	2	116
Weights, lb			
Engine	4600	792	1500
Stage dry weight	9385	10132	16941
Propellant	25394	53000	7347
Stage life, number of missions			
Engine	80	10	5
Structure	100	40	40
Basic costs, $M			
DDT&E[a]	1966	994	2074
Engine	28.8	6	40
Stages (first two)	161.8	101.9	198.2
Operations (first 20 missions)	596.4	879.9	628.15
Life cycle costs, $B			
20 missions	2.72	1.98	2.9
100 missions	4.97	5.36	5.3
500 missions	15.9	22.0	16.9
1000 missions	29.4	42.6	31.3
1542 missions (Model A)	43.8	64.8	46.7
2972 missions (Model C)	81.8	123.2	87.4
3669 missions (Model B)	100.2	151.6	107.0

a. DDT&E - Design, Development, Test, and Evaluation.

ascertain their propellant needs for Lunar missions. For operational systems, as indicated earlier, the propellant-in-orbit costs are the major operational cost and provide a basis for assessing the cost attractiveness of the competing propulsion systems, even though the LCC analysis is not extended to the Lunar missions.

2.4.1 Groundrules and Assumptions

The Lunar mission developed is compatible with delivery of manned or unmanned payloads to and from the Lunar surface. The mission assumes delivery of 40,000 lb of payload to the Lunar surface with 20,000 lb returned to LEO. The OTV is completely reusable.

This analysis assumes the use of the space station in LEO (220 NM and a Lunar station/platform (100 NM). For use of Lunar LOX, a Lunar base is assumed to be already established with mining and refining capabilities. Aerobraking at Earth is assumed where noted. Table 20 gives the vehicle sizing rules as applied to the Lunar mission. No restrictions at this point are assumed about the use of nuclear reactors for aero-entry at Earth or for manned Lunar-Trans-Lunar vehicles. For all comparisons, a platform or station in low Lunar orbit (LLO) is assumed as a transportation node for propellant transfer and LDAV (Lunar Descent/Ascent Vehicle) basing. the LDAV is assumed to be fully reusable. No resupply or ΔV capability is provided to support the Lunar platform/station in the analysis.

2.4.2 Transportation Nodes/Networks

Three nodes are employed to support a Lunar base: the space station, a Lunar space station/platform, and the Lunar base itself. Figure 19 shows the velocity changes necessary in Earth-Moon space.

Cycling orbits were dropped from consideration for Lunar base support because of the short flight time, and the fact that a periodic impulsive delta-V is needed for each Lunar fly-by to keep the CASTLE (Cycling Astronautical Spaceships for Transplanetary Long-Duration Excursions) in a cycling orbit. The combination of large weight and periodic delta-V makes CASTLES unattractive for the moon.

The nuclear electric vehicle uses the same nodes as the impulsive rockets but requires much higher ΔV's due to the spiral/gravity losses incurred.

TABLE 20. LUNAR OTV AND LANDER ASSUMPTIONS

CHEMICAL AND NUCLEAR OTV	
Tankage and unused propellant	6% of propellants used and fluids carried as payload
Aerobrake	15% of vehicle weight at atmosphere entry
Chemical engine thrust/weight ratio	50 to 1
Nuclear engine thrust/weight ratio	15,000 lbf per 6000 LBM engine
Structural weight	2% of maximum vehicle weight including payload
Minimum acceleration	0.1 g
Engine Isp	480 sec chemical-950 sec nuclear
NUCLEAR ELECTRIC OTV	
Power system and structure	16,500 lbm
Tankage and unused propellant	6% of propellants used and fluids carried as payload
Thruster power	500 kW
Thruster Isp	4000 sec
CHEMICAL AND NUCLEAR LANDER	
Tankage and unused propellant	10% of propellants used and fluids carried
Aerobrake and parachute	15% of vehicle weight at atmosphere entry
Chemical engine thrust/weight ratio	50 to 1
Nuclear engine thrust/weight ratio	15,000 lbf per 6000 lbm engine
Structural weight	5% of maximum vehicle weight including payload
Minimum acceleration	1.5 g's minimum at takeoff
Engine Isp	480 sec chemical-950 sec nuclear

is transferred to the LDAV to allow a Lunar landing and an ascent with 20 Klb of payload. The 20 Klb payload is attached to the OTV and the return flight made to LEO (with or without an aerobrake) for rendezvous with the space station. The OTV tanks are empty at arrival.

The ratio of propellant required for payload (40 Klb) delivered to the Lunar surface ranges from a maximum of 3.6 to a minimum of 1.8. The aerobrake weighs 5800 lb and, if needed to be replaced often, would increase the 1.8 ratio, which is for use of an aerobrake.

The use of a nuclear propulsion LDAV is thought to be very reasonable. Although the thrust to weight ratio is only average, the low Lunar gravity offsets this to a large extent. If an accident were to occur with the LDAV in the Lunar environment the spread of contamination would be small because of lack of an atmosphere. Disposal of a nuclear lander at the end of its life would be by permanent storage on a remote area of the Lunar surface. Figure 20 shows a concept for a nuclear propulsion LDAV. The vehicle would be designed for robotic operation; however, delivery of manned payloads would be part of the capability.

2.4.4 Chemical Propulsion Scenario

The chemical OTV Lunar mission scenario is very similar to the nuclear OTV scenario if Earth-based propellants are used. The velocity requirements are the same as the nuclear OTV and are shown in Figure 19. Ratio of propellant required in LEO to payload delivered to the Lunar surface is 6.1. The previous comments regarding the reusability of the aerobrake also applies to the chemical OTV.

The production of LOX from Lunar surface material has been considered. The oxygen produced would be used for the personnel living on the Lunar surface, sent to LEO for Space Station use and used as a propellant. The use of Lunar LOX for propulsion has been analyzed. The chemical OTV would leave LEO as in previous mission profiles loaded with enough LH_2 for the round trip of the OTV and a round trip of the LDAV (a total of 64.4 Klb). It would carry only enough LOX for the transfer

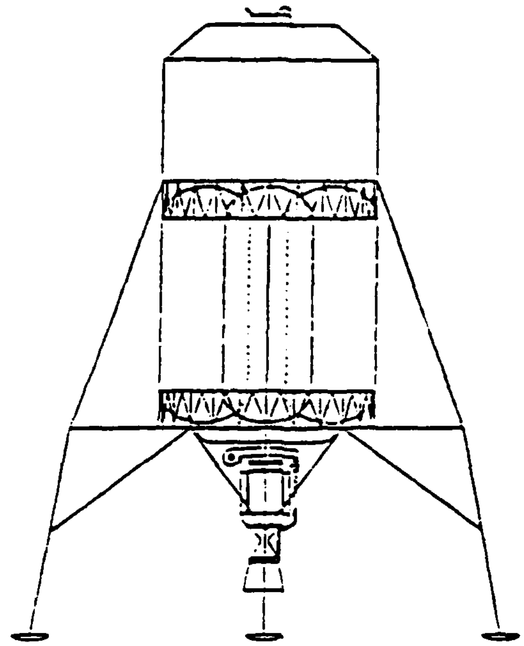

Figure 20. Lunar descent/ascent vehicle using nuclear engine.

burn to Lunar orbit. The LOX would have been transferred to LEO from a previous trip to the Moon. The 40-Klb payload would be transferred to the LDAV as well as enough LH_2 for a descent and ascent. The LDAV in Lunar orbit would have LOX for a descent trip as well as excess to transfer to the OTV for its return trip to LEO and the first leg transfer on a future trip.

The ratio of LH_2 required in LEO to payload delivered to the Lunar surface is 1.8. This is a very favorable ratio; however, total Lunar LOX production for each mission is estimated at 451.0 Klb. Total LOX transported to Lunar orbit is 211.2 Klb. The difference is the LOX used by the LDAV. If the quantity of LOX in Lunar orbit is added to LH_2 in LEO, the result is 243.8 Klb. The propellant to payload ratio then becomes 6.9. The production cost of Lunar LOX is very high and there may be a cost associated with transportation of LOX to Lunar orbit. Other costs associated with Lunar LOX are summarized in Table 21.

The return of a chemical OTV to LEO with a load of Lunar LOX requires a massive aerobrake (36,000 lb). If this aerobrake needs to be replaced often, the mass would need to be transported to LEO and have a negative impact on the propellant-to-payload ratio. Production of aerobrakes at the Lunar base could be considered as an alternative.

TABLE 21. NEGATIVE IMPACTS OF LUNAR LOX FOR USE AS OTV/LDAV PROPELLANT

Cost of Lunar LOX production facility (DD&T, transportation to moon, setup cost)
Support for Lunar LOX plant (extra personnel, equipment, consumables, earth support)
Very large LDAV required (57 Klb dry, 538 Klb loaded)
Contamination of Lunar environment with large LDAV operations
Design of massive OTV aerobrake required
LOX returned to LEO may need to be stored for a long period of time before next OTV flight

2.4.5 Electric Propulsion Scenario

The mission profile for the electric propulsion OTV is quite different from the high thrust chemical or nuclear stages. Low thrust requires the electric OTV to spiral out from LEO to the vicinity of the Moon where it then spirals in to the 100-Nm Lunar space station orbit. This transfer is not as efficient as the high thrust transfer, as shown in Figure 21. The velocity requirement each way is 26,000 ft/sec. Since this profile uses Earth-based propellants, the electric OTV needs to carry the propellant required by the LDAV.

Figure 22 shows the relationship between transfer time and reactor power level for this mission to Lunar orbit. It is reasonable to assume that even at relatively high power levels this transfer time is too great for a manned mission. The increased size of the habitat and increased

consumables will offset the savings in propellant from a high Isp. The use of an aerobrake is not considered reasonable for an electric stage because of the size of the stage and the lightweight structure.

The 500-kWe electric OTV sized for the GEO mission was used to calculate propellant ratios. The ratio ranges from 1.3 to 2.2. These very favorable ratios are associated with long transfer times and could likely only be used for transfer of cargo.

2.4.6 Summary Comparison of the Competing Propulsion Systems

Table 22 contains a summary of all Lunar missions analyzed. Since its long transfer time disqualifies the electric propulsion OTV for a manned mission, the nuclear OTV is the clear winner. Comparing nuclear and chemical options, propellant savings to LEO and LLO range from 98.2 to 203.2 Klb. At 750 $/lb to LEO or LLO, this results in a cost savings of 73.6 to $152.4M per flight.

When considering nontime critical delivery of payloads, the electric propulsion OTV, with chemical propulsion LDAV, has a propellant savings of 153.8 to 186.0 Klb over the chemical OTV. The corresponding propellant cost savings are 115 to $139M. With a nuclear LDAV, the savings increase an additional $28.9M per flight.

The most likely options from a technical and safety viewpoint would be chemical OTV with Earth-based propellants; nuclear OTV (WO/AB) and nuclear LDAV; electric OTV with chemical LDAV (if a nuclear engine were developed for a LDAV it would likely be produced for the OTV as well). With these assumptions the nuclear OTV and electric propulsion OTV have savings of $113.4M and $115.3M, respectively, over the chemical option. Although the electric OTV option has a slight edge, the nuclear OTV is much more flexible due to shorter transfer time.

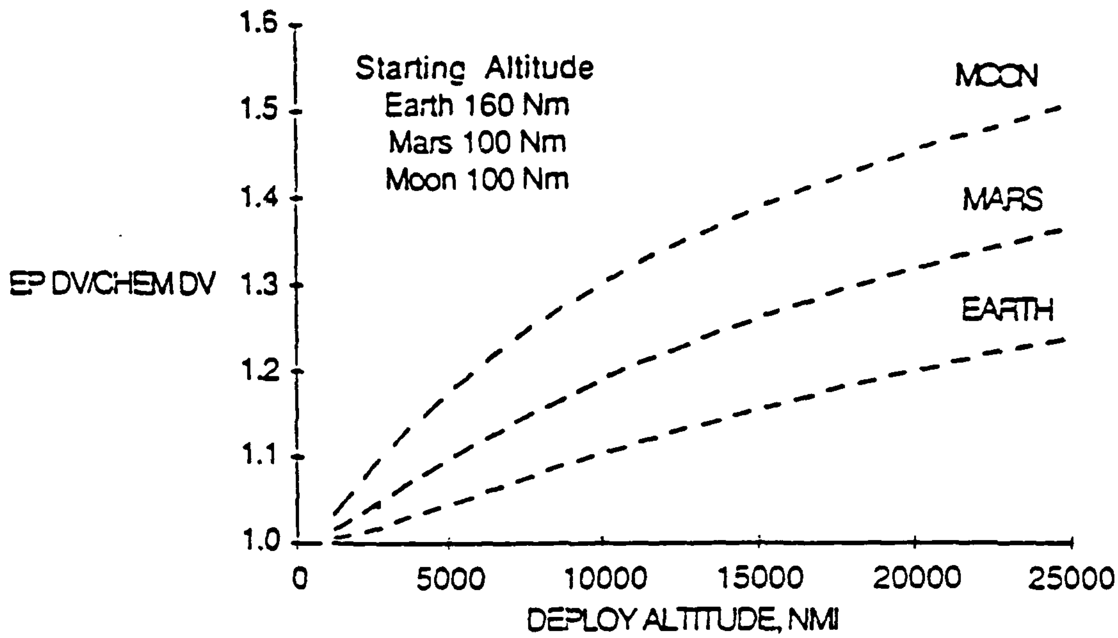

Figure 21. Electric/chemical ΔV ratio vs. deploy altitude.

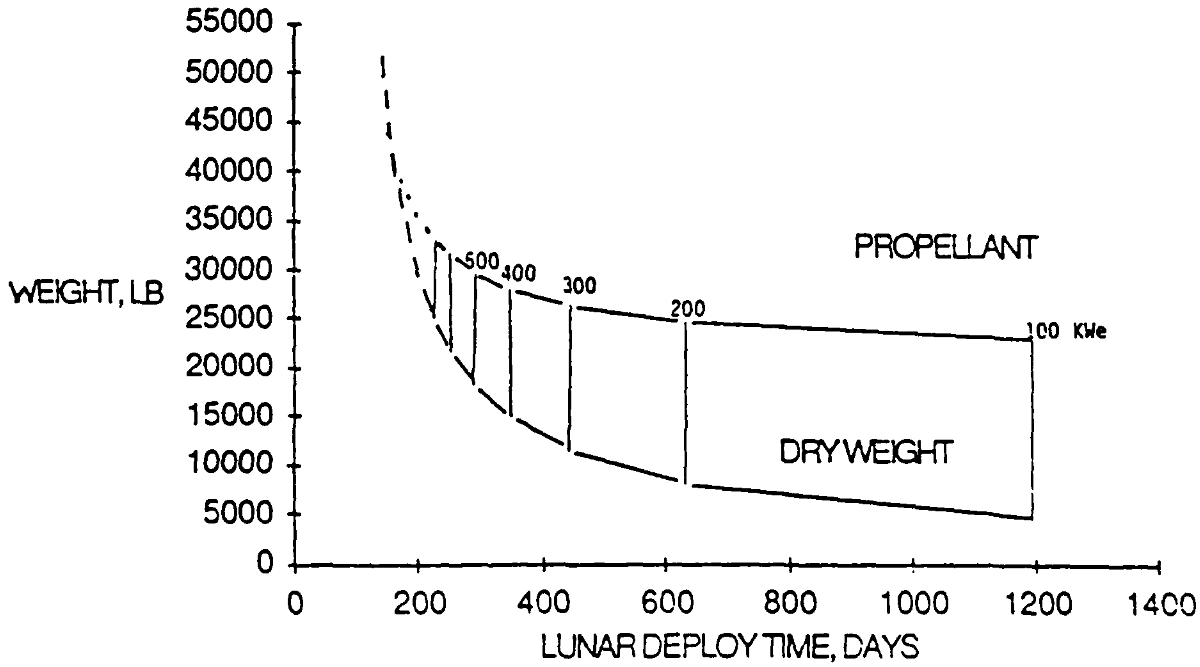

Figure 22. NEP Lunar mission dry and propellant weight vs. deploy time.

TABLE 22. LUNAR MISSION PROPELLANT COMPARISON

	OTV Prop (Klb)	LDAV Prop (Klb)	OTV+LDAV Prop (Klb)	LEO Prop (Klb)	LEO Prop P/L(2)	LEO+LLO Prop P/L(2)
Nuclear OTV						
W/AB, Nuclear LDAV	50.5	22.3	72.8	72.8	1.8	1.8
WO/AB, Nuclear LDAV	70.3	22.3	92.6	92.6	2.3	2.3
WO/AB, Chemical LDAV	92.8	52.8	145.6	145.6	3.6	3.6
Electric OTV (WO/AB)						
Nuclear LDAV	29.2	22.3	51.5	51.5	1.3	1.3
Chemical LDAV	37.2	52.8	90.0	90.0	2.2	2.2
Chemical OTV (W/AB) & LDAV						
Earth Based Propellant	191.0	52.8	243.8	243.8	6.1	6.1
Lunar LOX (1)	273.9	241.5	515.4	64.7	1.6	6.9

Assumptions:

40K lb payload delivered to Lunar surface
20K lb payload returned from Lunar surface to LEO

Nuclear Isp 950 sec
Electric Isp 4000 sec
Chemical Isp 480 sec

LDAV tank mass + unusable propellant = 0.1 x propellant mass
OTV tank mass + unusable propellant = 0.06 x propellant mass

(1) 451.0 Klb Lunar LOX production with 211.3 Klb delivered to LLO
(2) 40 Klb delivered to Lunar surface

AB Aerobrake
LDAV Lunar Descent Ascent Vehicle
LEO Low Earth Orbit
LLO Low Lunar Orbit
P/L Payload

2.5. Mars Base Support Missions

Several scenarios, with their associated assumptions, have been advanced for travel to Mars. These missions can be grouped into three broad classes: one, send an exploratory manned mission to "plant the flag" in the same vein as Apollo. Two, send vehicles periodically to establish and supply the base continuously. Three, establish transportation systems capable of economic support of a Mars base with cycling transport ships (CASTLEs), a Mars space station, and reusable Mars descent/ascent vehicles.

This study examines the use of nuclear electric, nuclear thermal, and conventional chemical/cryogenic propulsion schemes to supply an established Mars base with resupply, crew exchange, and processed material return to

Earth. Each of the three propulsion systems is matched with its appropriate trajectories and vehicles.

2.5.1 Ground Rules and Assumptions

This section describes the vehicle sizing assumptions and mission ground rules used to provide a comparison between engine technologies.

Each architecture must carry 40,000 lb of payload from LEO (220 Nmi, space station orbit) to the Martian surface and return 20,000 lb to the space station at Earth. Flight time is not constrained and it is assumed that the payload is not diminished in value or quantity over time. All propellants originate from the Earth and are initially loaded on the vehicle in LEO. All propellant and cost figures assume an established transportation architecture.

For nuclear electric a minimum total burn time is assumed for the spiral trajectories. The reactor is sized at 500 kW electrical power and MagnetoPlasmaDynamic (MPD) thrusters are used at a specific impulse of 4000 sec. The assumed specific weights are 10 kg/kWe for the power system and 5 kg/kWe for the engine system. The vehicle sizing assumptions for the nuclear electric system are the same as for the Lunar mission (see Table 20). Tankage mass is 6% of both the propellant needed by the MPD thrusters and that needed by the Mars Descent/Ascent Vehicle (MDAV). The chemical and nuclear rocket vehicles are sized in a similar manner. The only differences are in engine weight and Isp. The lander weights are more robust due to the higher G-loads encountered. G-levels listed are relative to the planet or moon for which the vehicle is used. Four vehicles used are fixed in weight; the CASTLE at 880,000 lb, the Mars space station at 1,000,000 lb, and the landers at 114,000 and 146,000 lb for nuclear and chemical, respectively. The propellant needed is 48,000 and 98,000 lb for the nuclear and chemical landers, respectively. These propellant loads must be supplied by the OTV/Taxi to satisfy the terrestrial propellant groundrule.

2.5.2 Transportation Nodes/Networks

This section describes the trajectories used and their corresponding velocity changes. Each method of transfer from Earth to Mars and return has different velocity requirements. Much of this work is based on reports from JPL and SAIC, as listed in References 1 and 2.

Ballistic, minimum energy transfers have been, to date, the only physically possible method of transfer to Mars (Figure 23 shows a typical trajectory). Although the velocity requirements vary with each opportunity, average values can be used. The injection energy, or C_3, used is 13 km^2/s^2 which translates to 12,400 ft/s ΔV from a space station orbit assuming no plane change. At Mars, the infinite approach speed is 3.0 km/s, which translates to 7500 ft/s ΔV for the capture burn into a circular 100 Nmi orbit. If aerobraking is used, then the ΔV is based only on circularization, which for Earth is 310 ft/s and for Mars is 118 ft/s. For the return trip to Earth, these same numbers are valid.

For low thrust, spiral trajectories the velocity requirements are much higher. Based on JPL's analysis of a single vehicle manned Mars mission using a 3 MW jet power level and a 290 metric ton departure weight at GEO, the one way ΔV to a 3000 km orbit about Mars is 62,000 ft/s. Figure 24 shows the low thrust trajectory.

Cycling orbits are orbits that have repeated fly-bys of the Earth and some target body, in this case Mars. By placing a large vehicle capable of supporting a crew of seven with consumables and comfortable living quarters into a cycling orbit, only small vehicles are needed to transfer the crew from the CASTLE (Figure 25) to the space station circling each planet. This avoids costly energy demands of large support vehicles.

The cycling orbits have several ΔVs associated with them. Figures 26 and 27 and Table 23 show the transportation nodes and the associated velocity increments needed. All of the numbers represent average values and are based on the VISIT-1 cycling orbit (Figure 28).

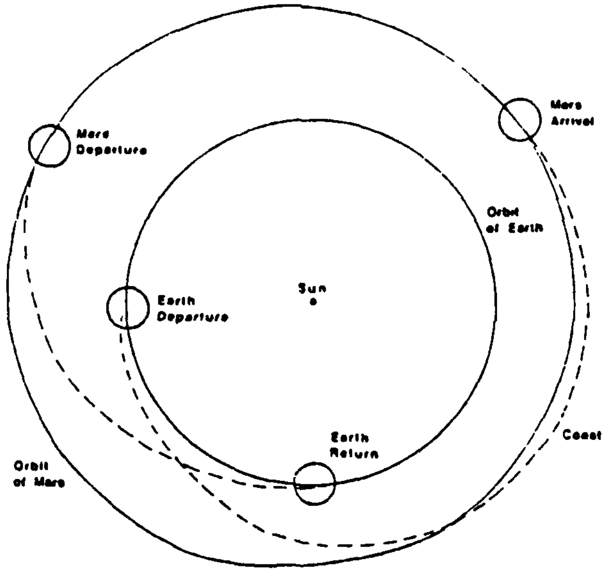

Figure 23. Mars direct ballistic trajectory.

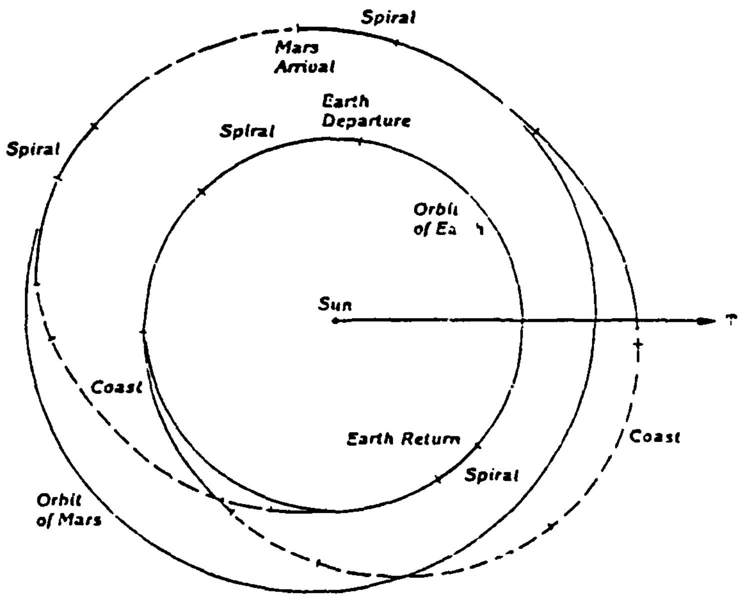

Figure 24. NEP interplanetary trajectory profile.

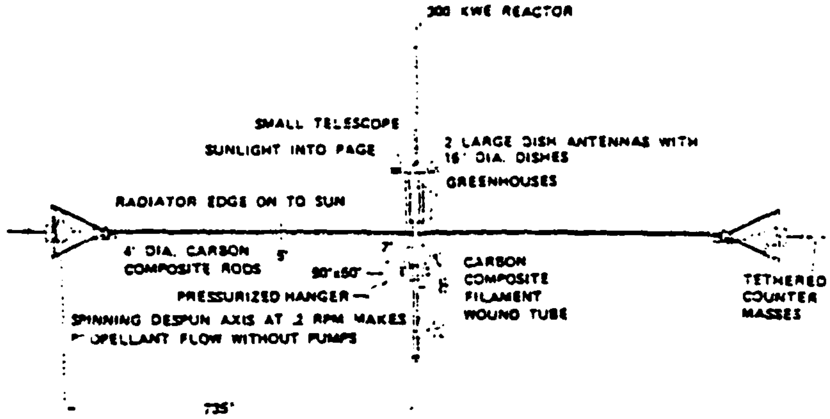

Figure 25. CASTLE conceptual design.

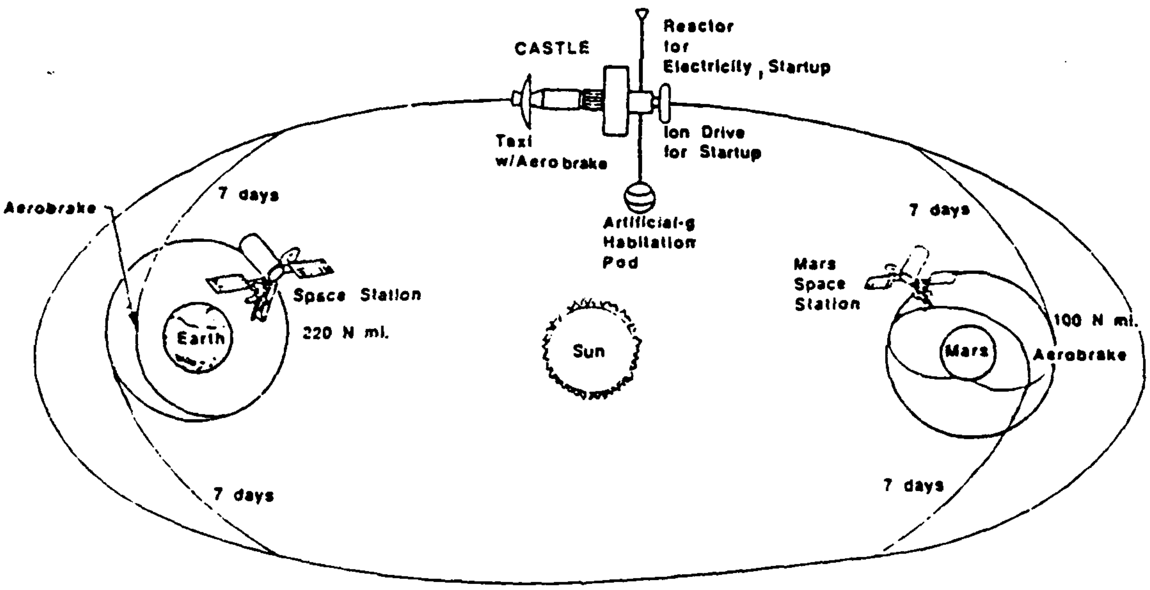

Figure 26. Mars transportation nodes using CASTLEs.

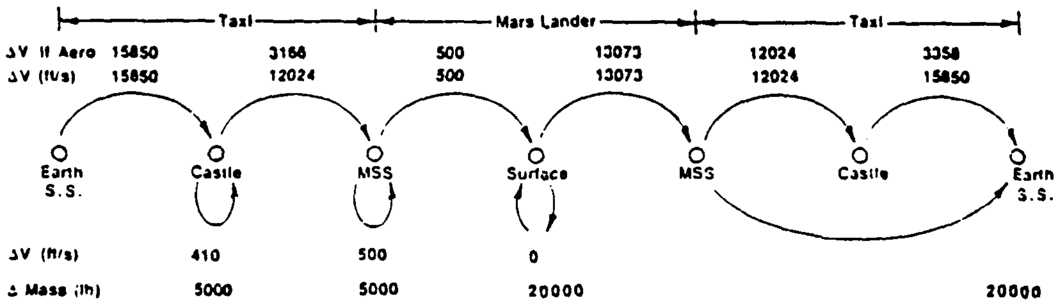

Figure 27. Mars transportation velocities using CASTLEs.

TABLE 23. CYCLING ORBIT VELOCITY REQUIREMENTS

Transportation Node	Δ V (ft/s)
LEO-CASTLE	12,802
LMO-CASTLE	8,976
CASTLE (TCM)	410
Mars Base (OTM)	500
LMO-Mars Base	500
Mars Base-LMO	13,073
CASTLE Intercept-LEO (Aero)	310
CASTLE Intercept-LMO (Aero)	118
Taxi deflection/alignment	3,048

LEO Low Earth Orbit (220 Nmi)
LMO: Low Mars Orbit (100 Nmi)
CASTLE: Cycling Astronautical Spaceships for Transplanetary Long-duration Excursions
TCM: Trajectory Correction Maneuver
OTM: Orbit Trim Maneuver

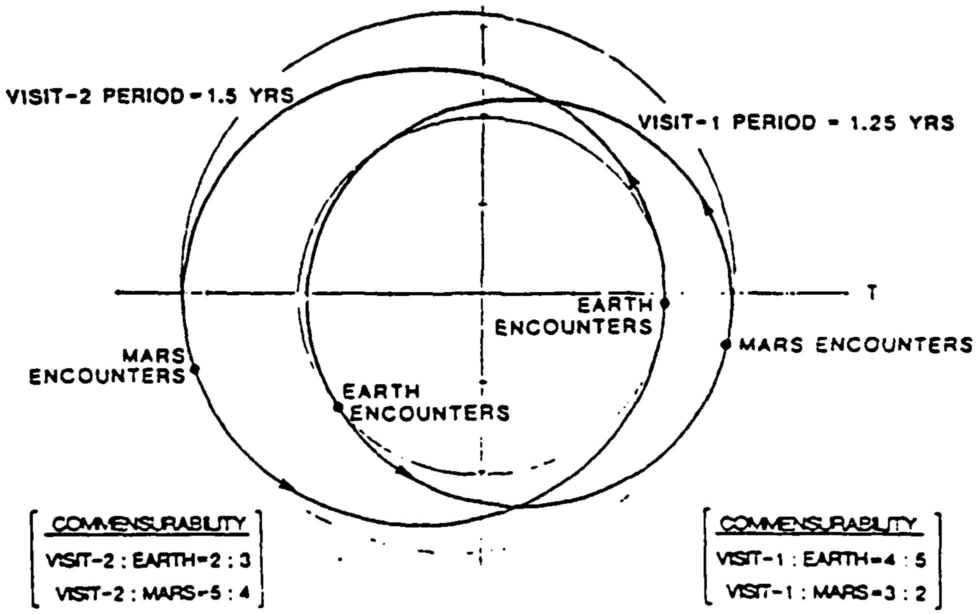

Figure 28. VISIT cycling orbits.

Following is a chronological profile of a trip from Earth to Mars and back again using a cycling orbit architecture. A traveler would first arrive at Earth's space station, then transfer into a taxi/tanker which carries the crew and all propellants needed for the upcoming round trip mission to Mars. The taxi performs its burnout of Earth orbit to begin an intercept course with the CASTLE. After one week the taxi rendezvous with the CASTLE and performs another burn to bring its course tangent with the CASTLE. Because each fly-by of the CASTLE is at a different distance, each taxi ride has a different cruise time and velocity requirement. The numbers in Table 23 are averages. Once aboard the CASTLE, the taxi transfers propellant and payload to the CASTLE for TCMs and resupply later in its mission. After several months of cruise in the CASTLE, the crew once again enters the taxi and departs for the Mars space station. A large burn is executed to deflect the taxi onto a Mars intercept course. The taxi either uses aerobrakes or engine thrust to provide capture at Mars. The crew and a large amount of propellant transfer to the MDAV for the trip to the surface. The Mars space station is resupplied with propellant and supplies as was the CASTLE. The CASTLE TCM (Trajectory Correction Maneuver) and Mars Base OTM are performed when a taxi is not attached. The same engine type as the taxi is assumed for these maneuvers. The MDAV descent is achieved with a retro burn to deorbit followed by an aerobraked entry. At the appropriate altitude, parachutes are deployed and a final retro burn provides a soft landing. Once on the surface, 40,000 lb (including the crew) is off-loaded for the base: 20,000 lb is on-loaded for the return trip. The launch must wait several months until another CASTLE approaches. Ascent is achieved with conventional engines. The Mars Base to LMO includes a 15% additional penalty to account for gravity, pressure, and other losses associated with planetary rocket ascent. No mass transfer to the Mars space station or CASTLE occur on the return leg. The crew and payload are transferred to the taxi and the taxi departs for an intercept with the oncoming CASTLE. The same rendezvous techniques are needed for the return trip. The last leg is an aerobrake entry into Earth's atmosphere and final rendezvous with the space station.

2.5.3 Nuclear and Chemical Propulsion Scenario

The OTV and lander assumptions, which are the same as for the Lunar mission, are shown in Table 20. The velocity requirements for the ballistic transfer are given at the beginning of the previous section and those for cycling orbits are shown in Table 23. Both the chemical and nuclear OTVs use Earth-based propellants.

The ballistic mission begins in LEO and the OTV carries enough propellant to make a round trip for the OTV and the MDAV. The mission is very similar to the Lunar mission using Earth-based propellants. The primary difference is the use of an aerobrake on the OTV at both Earth and Mars capture. The MDAV uses aerobraking/parachute for landing. Table 24 gives the MDAV weights for nuclear and chemical propulsion. Tables 25 through 28 show weight breakdowns for the OTV/taxi designs for various options for use of nuclear engines with nuclear or chemical landers and with and without aerobrakes.

2.5.4 Electric Propulsion Scenario

The low thrust of the electric propulsion OTV requires the vehicle to spiral out from LEO, transfer to the vicinity of Mars and do a slow spiral to low Mars orbit. The slow spiral trajectory is very inefficient, as shown in Figure 21. This results in a mission ΔV of 124,000 ft/s round trip. This large velocity offsets much of the advantage of the high Isp.

The mission was analyzed using the 500-kWe power system sized for the GEO and Lunar missions. This results in round trip times of 1600 to 2200 days. This indicates that the mission needs to be performed with a much larger power system. However, the propellant numbers obtained are representative of a electric OTV delivery system. An increase in the power system by a factor of two would not increase the propellant usage by a great amount since power system weight is a smaller fraction of total vehicle weight compared to the GEO or Lunar vehicles.

TABLE 24. WEIGHT BREAKDOWN FOR MARS DESCENT AND ASCENT VEHICLE

	Nuclear	Chemical
Takeoff Propellant (Klb)	45.7	91.7
Landing Propellant (Klb)	2.5	5.9
Total Propellant (Klb)	48.2	97.6
Number of Engines	5	3
Engine Weight (Klb)	30.0	1.8
Tank (Klb)	4.8	9.8
Aerobrake (Klb)	23.1	28.0
Structure (Klb)	7.7	9.3
Dry Weight (Klb)	65.6	48.9
Gross Weight (Klb)	113.8	146.5
Payload at Takeoff (Klb)	20.0	20.0
Takeoff Weight (Klb)	131.3	160.6
Required Takeoff Thrust (Klbf)	74.8	91.5
Thrust at Takeoff (Klbf)	75.0	105.0
Thrust to Weight at Takeoff	1.5	1.7
Payload at Landing (Klb)	40.0	40.0

TABLE 25. TAXI WEIGHT BREAKDOWN FOR NUCLEAR STAGE (WITH AEROBRAKE) USING A NUCLEAR LANDER

Total Payload (Klb)	50.0
Propellant (Klb)	427.8
Number of Engines	4
Engines (Klb)	24.0
Tank (Klb)	25.7
Aerobrake (Klb)	44.7
Structure (Klb)	11.6
Dry Weight (Klb)	106.0
Gross Weight w P/L (Klb)	583.8
Required Thrust (Klbf)	58.4
Thrust (Klbf)	60.0
Thrust to Weight	0.10

TABLE 26. TAXI WEIGHT BREAKDOWN FOR NUCLEAR STAGE (WITH AEROBRAKE) USING A CHEMICAL LANDER

Total Payload (Klb)	50.0
Propellant (Klb)	618.6
Number of Engines	6
Engines (Klb)	36.0
Tank (Klb)	37.1
Aerobrake (Klb)	63.9
Structure (Klb)	16.4
Dry Weight (Klb)	153.4
Gross Weight with P/L (Klb)	822.0
Required Thrust (Klbf)	82.2
Thrust (Klbf)	90.0
Thrust to Weight	0.11

TABLE 27. TAXI WEIGHT BREAKDOWN FOR NUCLEAR STAGE (WITHOUT AEROBRAKE) USING A NUCLEAR LANDER

Total Payload (Klb)	50.0
Propellant (Klb)	1016.3
Number of Engines	8
Engines (Klb)	48.0
Tank (Klb)	61.0
Aerobrake (Klb)	0
Structure (Klb)	24.0
Dry Weight (Klb)	133.0
Gross Weight with P/L (Klb)	1199.3
Required Thrust (Klbf)	119.9
Thrust (Klbf)	120.0
Thrust to Weight	0.10

TABLE 28. TAXI WEIGHT BREAKDOWN FOR NUCLEAR STAGE (WITHOUT AEROBRAKE) USING A CHEMICAL LANDER

Total Payload (Klb)	50.0
Propellant (Klb)	1395.9
Number of Engines	11
Engines (Klb)	66.0
Tank (Klb)	83.7
Aerobrake (Klb)	0
Structure (Klb)	32.6
Dry Weight (Klb)	182.3
Gross Weight with P/L (Klb)	1628.2
Required Thrust (Klbf)	162.8
Thrust (Klbf)	165.0
Thrust to Weight	0.10

The propellant results are compared to the chemical and nuclear system numbers in Table 29. They are much lower than the chemical results and fall in the middle of the nuclear OTV results. The electrical system could use a chemical system for boost out of LEO to decrease the spiral inefficiencies and decrease transfer time. This would tend to increase the propellant usage to that of the worst-case nuclear rocket propellant usage numbers.

2.5.5 Summary Comparison of Competing Propulsion Options

The propellant usage for all missions analyzed in this study are summarized in Table 29. The propellant-to-payload ratio ranges from a low of 3.3 to 34.9. Of the three propulsion options considered for Mars missions, the ANRE has distinct advantages over both NEP and chemical/cryogenic. If cycling orbits and terrestrial propellants are baselined, then ANRE is the only propulsion system that can be used. NEP does not have the high accelerations needed for the taxis and chemical's Isp and mass ratios are too low for the large ΔV's needed, even with aerobrakes.

TABLE 29. SUMMARY COMPARISON OF MARS SCENARIOS

	MDAV	Propellants (Klb) OTV/ Taxi	MDAV+OTV/ Taxi	Propellant Ratio+*
Electrical Spiral				
Nuclear Lander-Electric	48	117	165	4.1
Chemical Lander-Electric	98	157	255	6.4
Direct Ballistic				
Nuclear Lander-Nuclear with AB	48	83	131	3.3
Chemical Lander-Nuclear with AB	98	116	214	5.4
Nuclear Lander-Nuclear	48	206	254	6.4
Chemical Lander-Nuclear	98	278	376	9.4
Chemical Lander-Chemical with AB	98	460	558	14.0
CASTLE/Cycling Orbits				
Nuclear Lander-Nuclear with AB	48	380	428	10.7
Chemical Lander-Nuclear with AB	98	521	619	15.5
Nuclear Lander-Nuclear	48	968	1016	25.4
Chemical Lander-Nuclear	98	1298	1396	34.9

+ (40,000 lb payload from LEO to Mars Surface)
(20,000 lb payload from Mars Surface to LEO)
* (Total propellant needed in LEO divided by payload weight)

All propellant Earth based
Nuclear Isp 950 sec
Chemical Isp 480 sec
Electric Isp 4000 sec

AB = Aerobrake
MDAV = Martial Descent/Ascent Vehicle
CASTLE = Cycling Astronautical Spaceships for Transplanetary Long-duration Excursions

Chemical propulsion can only be used in conjunction with aerobrakes and a direct ballistic trajectory. Without aerobrakes, the mass ratios are too low, and the mission is physically impossible. (Aerobraking cuts propellant requirements in half.) Table 29 shows that for the baseline mission of 40 Klb delivered and 20 Klb returned, a chemical/aerobrake system requires 558,000 lb of propellant. This gives a propellant to delivered payload ratio of 14.0. By comparison, a nuclear OTV without an aerobrake using a chemical lander at Mars uses only two-thirds the

propellant required by the chemical system. If an aerobrake can be used by the nuclear OTV at Mars and Earth, the propellant savings over chemical is 60%. If a nuclear Mars lander is used as well, then the savings becomes 75%, requiring only 131,000 lb of propellant per mission. Both chemical and nuclear systems would have the same time of flight to and from Mars.

NEP can compete in terms of propellant used; however, the savings is only 30% over a nuclear system without aerobrakes. If the nuclear system can use aerobrakes, then it uses 15% less propellant than NEP. This assumes both systems use chemical landers and that the ANRE flies direct ballistic trajectories. When time of flight is considered, ANRE becomes the clear winner. For ANRE it takes 200 days (Type-I, typical) one way; whereas, for the NEP system it takes 728 days one way. When crew consumables and mission risk are folded into the picture, the ANRE/ballistic trajectory becomes the preferred approach.

Further, if a nuclear lander is used in both the NEP and nuclear OTVs, then the ANRE uses 20% less propellant than NEP and has the lowest propellant requirement of all Mars missions analyzed.

2.6 Nuclear Stage Safety Considerations

Nuclear rocket research was carried on from the mid-1950s until 1973 (Section 1). The original purpose focused on propulsion for an intercontinental ballistic missile. However, it was quickly redirected towards manned Mars exploration. This emphasis was modified in the later development stages to address (a) transfer of men and/or materials from earth-orbit to lunar-orbit and return and (b) an injection stage for deep-space probes and return of the stage to Earth-orbit. The program was named Rover/NERVA (NERVA is an acronym for Nuclear Engine for Rocket Vehicle Applications). Part of the program included a significant effort to define and resolve the safety issues associated with nuclear rockets.

There is current renewed interest in an ANRE as an earth-orbit tug. Compared to manned missions to Mars, the "near" Earth space operations of current primary interest involve performance requirements that are

considerably less demanding. ANRE utilizes a single propellant (hydrogen) and therefore is inherently safer than chemical rocket engines which require dual propellants. The fuel and oxidizer of chemical rockets have the potential for unplanned combination and explosive ignition at any time during launch operations, space storage, or mission use. This possibility does not exist with the ANRE/NERVA type engines.

During launch, the core is not radioactive and any launch accident would result in the return of harmless fuel to earth. Water immersion of a full core, if poison rods are inserted in the core, will not result in criticality. During launch and storage, the single propellant nuclear engine eliminates the possibility of an explosive combination of propellant and oxidizer characteristic of a chemical stage, and lower component stress levels should provide more reliable operation.

For a tug, the thrust (and thus engine power) required is on the order of a fifth or less than that of the NERVA engine. Also, the orbital maneuvering operations, which constitute the majority of the missions applicable to space-based nuclear stages, require burn times in the range of 3-8 min. Though most servicing and repairs are expected to use robotic systems for these relatively short burn times, the engine does not become highly radioactive; and limited manual servicing of the engine system components above the core shadow shield is practical (Figure 29). Most of the engine system operating components (e.g., pumps, valves, actuators, etc.) are in this region. For example, the projected radiation levels in this area following a 5-min burn are 0.5 rad/hr and 0.03 rad/hr for 1 and 10 days after shutdown, respectively. Thus, individuals could provide about 10 hr and 150 hr, respectively, of manual service in this area ithout exceeding the present guideline limit of 5 rads per year for radiation workers.

By limiting the nuclear engine to short burns while in LOE and by establishing proper design criteria, the nuclear engine can be designed to avoid serious core disintegration after a loss of coolant accident from full power. Such procedures will protect the stage and permit subsequent removal of the engine to a safe orbit.

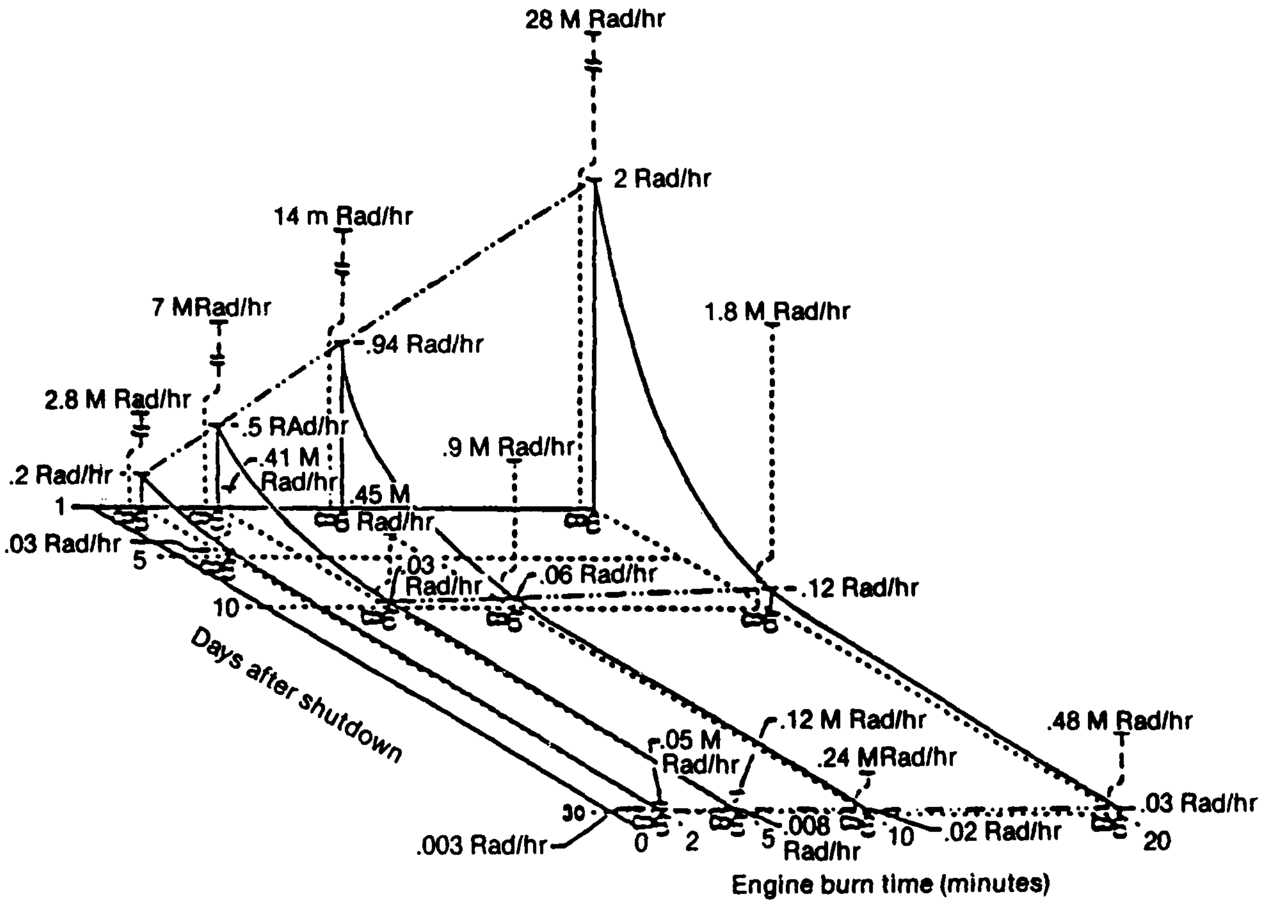

Figure 29. Nuclear engine radiation intensity--strength/decay.

Other possible applications for nuclear rockets, in addition to orbit transfer vehicles or tugs, are in planetary transfer, manned planetary missions, launch vehicles, upper stages, single-stage to orbits, and lunar launch vehicles. Ensuing discussions concentrate on the orbit transfer vehicle or tug and, when appropriate, introduce the issues in using nuclear rockets for other applications.

The major concerns relative to propulsion flight safety are the following:

1. Preventing unplanned nuclear criticality.

2. Providing for radiological safety in the case of random landing locations resulting from a launch vehicle abort.

3. Providing for safe reactor disposal or prevention of unpredictable reentry into the biosphere after operation.

4. Reducing to acceptable radioactivity levels the exhaust plume of vehicles using a single stage to orbit nuclear rocket engine.

2.6.1 Flight Safety Lessons from NERVA

Reliability and safety of the engine design were of paramount concern during all phases of the NERVA program. A major, high priority effort was directed toward eliminating from the engine design those single failures or credible combinations of errors and failures which could endanger mission completion, the flight crew, the launch crew, or the general public. Probabilistic design and failure mode and effects analysis were included in this effort. Examples of the effects of these analyses on flight engine design are the incorporation of redundant turbopumps and the use of four valves in place of each single valve. Where no practical engine design solutions were found for credible single or multiple failures that could jeopardize crew or population safety, appropriate countermeasures and alternative operating modes were explored. For example, provisions were made for engine operation in an emergency mode to effect safe crew return and to prevent danger to the Earth's population in the event a planned mission were to be abandoned because of engine failure.

Accident events were divided into two categories: (a) the casual accident or primary failure(s) and (b) the nuclear accident. The former was defined as the event or series of events which, if unimpeded, could culminate in the latter. The nuclear accident was defined as one involving

abnormal release of direct radiation and/or radioactive material into the biosphere. For example, the failure of the chemical booster rocket used to lift the nuclear rocket to its staging altitude could cause a loss of reactor control, and thus countermeasures were needed to avoid a nuclear event. The safety philosophy adopted was to assume the credible casual accident and potential nuclear accident. One example would be the use of an explosive charge if the booster malfunctioned to render the reactor permanently subcritical.

The operational philosophy adopted was to confine the powered operation of the nuclear rocket beyond the active biosphere. Hence, the nuclear rocket was an upper stage which was to be started in space rather than on the ground.

Operational missions envisioned included (a) preorbital startup, orbital parking, and restart to escape; (b) preorbital startup direct to escape; and (c) injection by chemical boosters into a parking orbit, with orbital startup to escape. The launch site was to be the Kennedy Space Center. The hydrological features downrange are quite favorable. Approximately 400 km downrange, a submarine cliff named the Blake Escarpment cuts the North American continental shelf. At the uprange edge of the escarpment, the water depth is about 1100 m; on the downrange side, it is several miles deep and remains at or near this depth for 8000 km or more downrange. The bottom of this deep basin presents a safe resting place for a nuclear rocket engine should in-flight failures of the nuclear stage occur. The interchange of surface to bottom water in this basin takes hundreds of years. Importantly, it is possible to maintain the flight of a nuclear rocket over water for distances up to and beyond the orbital injection point.

Projected operating modes of nuclear rockets include both suborbital and orbital startup of the nuclear stage. In successful missions, in which escape velocity is achieved, safe and final disposition of the spent engine and its radionuclides in space is effected. However, if nuclear stage propulsion or guidance failures yield velocity decrements short of orbital

or escape velocity, there is a potential problem of reentry of the nuclear rocket into the Earth's atmosphere at an unpredictable time and location.

Suborbital start-up failures occurring in the 4300 m/s (14,000 ft/s) to 7600 m/s (25,000 ft/s) velocity band were divided into two categories: (a) early failures yielding ballistic reentry and (b) late failures resulting in reentry from partial, low altitude, or highly eccentric orbits. The early failures result in velocity decrements, generally 300 m/s (1000 ft/s) or more short of orbital velocity, and would cause the nuclear stage to impact in deep water within the limits of the missile range. Late nuclear stage guidance or propulsion failures could inject the nuclear rocket into orbital flight paths of variable lifetimes. Regressions from these flight paths were either prompt (minutes or hours) or delayed (days or months), but the subsequent reentry point could be outside the limits of the missile range. The reentry event was also random, and it was quite difficult to control the time and place of reentry.[3]

In the case of orbital start-up or orbital restart to escape, a different situation exists. It was concluded that early thrust cut-offs of a nuclear rocket with proper guidance control would result in orbital lifetimes which are always greater than the lifetimes of the parking orbit. In cases where the planned thrust program was executed, the angle of thrust application varied approximately ±40 degrees and still yielded Earth escape velocity conditions. More acute guidance failures could yield low perigee elliptical orbits of short lifetimes or hyperbolic flight paths resulting in prompt random reentry. Since only acute guidance failures could result in prompt reentry of the engine with its full radionuclide inventory, a guidance-propulsion interlock, to prevent or terminate thrust if attitude control failed, would be an effective safety device. As a preventive measure, increasing the perigee and lifetime of parking orbits would also be effective since it increases the time available for radioactive decay of fission products within the engine.[3]

NERVA Safety Features

Accidental insertions of reactivity could occur from (a) a control system malfunction that (b) floods the reactor core with water or impaction. The energy release, if an accident supercritical condition occurred, depends upon (a) the amount of reactivity inserted, (b) the rate of insertion, (c) the initial state of the reactor (e.g., hot or cold), and (d) the quenching or shutdown mechanism. Rapid insertions of large amounts of reactivity would be accompanied by releases of kinetic energy which physically disrupt the reactor. A test called KIWI-TNT was conducted to demonstrate the effects of large and rapid reactivity insertion. Special actuators were used to achieve the desired reactivity rates. The excursion released 10,000 MW(s) of energy and completely dismantled the core in a mechanical (not nuclear) explosion.

The planned nuclear rocket engine stage was a modified Saturn vehicle with the nuclear upper stage replacing the S-IV B. The potential energy releases of the booster propellants as a result of booster failure was a predominant factor in range safety. The Saturn booster fueled with liquid oxygen and RP-1 included 2,180,000 kg (4,800,000 lb) of propellants and the S-II stage fueled with liquid oxygen and liquid hydrogen included 386,000 kg (850,000 lb) of propellants. In case of a destruct, it was calculated that 10% of the Saturn booster and 60% of the S II stage kinetic energy, or the equivalent of 218,000 kg (480,000 lb) of TNT from the former and 231,000 kg (510,000 lb) of TNT from the latter, needed to be considered in kinetic energy release. The nuclear stage included a destruct system that was integrated with the booster destruct system. In addition, an engine destruct system would be tied to the nuclear stage destruct system. Therefore, if vehicle or nuclear stage destruct action was necessary, the reactor would also be rendered safe. An ordinance destruct system would fragment the reactor into particles small enough to remain aloft as aerosols to be burned up upon reentry into the Earth's atmosphere or with so little activity upon reaching the Earth's surface that they would not present a hazard. Conceptual destruct methods follow:

1. Thermochemical destruct--injection of reactive chemicals into the reactor core such as UF_6

2. Nuclear destruct--inducing nuclear transients within the core

3. Explosive destruct--chemical explosives (a destruct system consisting of four 52-in. 105-mm rounds containing Composition B explosives was designed and tested)

4. Metallurgical additives--placement of discrete metallurgical additives within the fuel to sensitize it to the destructive action of alternative methods.

The transfer of nuclear rockets in the space shuttle would probably eliminate the need for a destruct system.

The development of neutron poison systems to "safe" the reactor during its transport to the missile test site, during ground handling, and possibly during the early stages of launch, was a primary thrust of the nuclear safety program. A redundant poison approach was pursued in which poisons could be inserted and reinserted into the core and reactor control elements could be locked. Therefore, if the control elements were withdrawn inadvertently, the core poisons could override the resultant reactivity insertion. Conversely, if the core poisons were withdrawn, the locked control system alone could save the reactor.

A number of advanced countermeasures were also considered. Propulsion guidance interlocks were considered to interlock the propulsion and guidance systems in a manner to activate thrust termination in the event of guidance failures during orbital start-up or restart to preclude prompt reentry. Retrosystems for inducing downrange impact in the event of late nuclear stage aborts during orbital injection to preclude random reentry were another idea. Also, retrosystems for inducing orbital departure and impact in predetermined marine disposal areas to counter random reentry were under investigation. Satellite interceptions might utilize ground-to-air or air-to-air missile systems to intercept and destroy

nuclear rocket reactors or induce their impact into predetermined marine disposal areas. Another idea considered was the use of auxiliary rockets to carry the nuclear rocket into orbit in case of late preorbital injection thrust failures or to transfer the nuclear stage to orbits of higher perigee in case of orbital start-up failures. This would provide additional decay time and also preclude prompt random reentry. Automatic malfunction sensors and countermeasure initiators using on-board malfunction sensors in the nuclear stage (to detect guidance, thrust, or propellant malfunctions connected to automatic on-board initiators which execute destruct or countermeasure action, if necessary) were also being evaluated.

The NERVA Safety Plan established many requirements for flight safety.[3] It stated, for example, that a maximum effort was to be directed toward eliminating from the engine design those single failures or credible combinations of errors and failures which could endanger mission completion, the flight crew, the launch crew, or the general public. If this effort proved impossible or resulted in an excessive penalty, redundancies internal to the component in question were to be considered. If this alternate approach also proved ineffective, ways in which other components could compensate were to be investigated. Where no practical solutions were found in inherent design and where credible single or multiple failures could jeopardize crew or population safety, countermeasures or techniques such as maintainability and alternative operating modes were to be explored. Further, if the planned mission was to be abandoned because of an engine failure, provisions were to be made for engine operation in an emergency mode to effect safe crew return and to prevent danger to the Earth's population. Operation in the emergency mode was to allow optimum use of remaining propellant commensurate with the failure and, at a minimum, provide engine performance on the order of 30,000-thrust and 500-sec specific impulse. In addition, the engine was to be capable of delivering a minimum controllable total impulse of 10^8 lb-sec including the impulse derived from the cooldown propellant. This total impulse was to be obtainable in a single thrust cycle with the powered-operation portion of the cycle at or above the specified thrust and specific-impulse minimums. This goal was to be obtainable from all

operating phases of the engine cycle, including all shutdown and coast phases. If engine failure occurred after the steady-stage powered phases of engine operation, provision was to be made for coolant up to 5 hr prior to entering the emergency mode. Final cooling was to preclude engine disintegration and (if possible at no additional risk to population, passengers, or crew) to preserve the engine in a restartable condition.

Additional NERVA safety design requirements were to have the engine incorporate the following features:

1. The means for preventing the inadvertent attainment of reactor criticality through any credible combination of failures, malfunctions, or operations during all ground, launch, flight, and space operations.

2. A destruct system during launch and ascent to ensure sufficient dispersion of the reactor fuel upon Earth impact to prevent nuclear criticality with the fuel fully immersed in water.

3. The means for preventing credible core vaporization or disintegration or violation of the thrust-load path to the payload.

4. Diagnostic instrumentation adequate to detect the approach of a failure or an event that could injure the crew or damage the spacecraft and the provisions to preclude such an event.

5. The capability for remote override of the engine programmer by the crew and ground control as well as for remote shutdown independent of the engine program.

6. An engine control system capability to preclude excessive or damaging deviations from programmed power and ramp rates.

Because of these safety concerns and the often indistinguishable relationship between safety and reliability, we have also reviewed the

NERVA reliability program. The reliability goal for the NERVA power plant was 0.995. This goal was in line with the NERVA design philosophy established by its director, Mr. Milton Klein:[4]

> The major design criteria for the NERVA engine development program shall be reliability and the achievement of the highest probability of mission success. Next in the order of importance must be performance as measured in terms of specific impulse. Then the engine design should attempt to keep the overall weight as low as possible within the bounds allowed by funds available for development. While there are interrelations between these criteria in design, I can see no basis for altering their order of importance.

Flight safety analysis was divided into three parts: malfunction analyses, fault tree analyses, and contingency analyses. Malfunction analyses were performed with a computer model and depict the system effects of the failure of components. Fault tree analysis is a deductive process by which an undesirable event is postulated and possible malfunctions which cause the event are systematically analyzed. Contingency analysis addresses component failures and how they are detected, system consequences of the failures, contingency actions required, and the time in which the contingency action must be performed.[5]

Analysis of component failures indicates a probability of about three failures per 1000 engine cycles for catastrophic failures. (Analysis was only performed on the nonnuclear engine components, but a review of the nuclear subsystem led to this number.)[5]

Designers had primary responsibility to prove that a component met specifications. The technique chosen to ensure that the reliability goal would be met was Failure Mode Analysis (FMA). FMA is a systematic method used to ensure that components have high, inherent reliability. The FMA developed for NERVA clearly defined the conditions for success. A probability equation was written to express each condition. This equation was then used to define the principal distributions and to provide an indication of the kind of analysis performed.

A thorough, unbiased narrative listing all credible ways failures can occur was written so that changes could be identified and used to eliminate

those failures or minimize their effects. This listing gave insight into fundamental causes and interactions and served as the basis of the subsequent reliability assessment.

The fundamental FMA steps are given below

1. Obtain the functional and physical description of the design to be analyzed.

2. Define the functional and physical boundaries, i.e., those items which will be included in this FMA as opposed to those which must be evaluated by other component or system FMAs.

3. Obtain or define probabilistically the input and output requirements.

4. List the component mode of failure, the operating conditions, the condition of success, the general design analysis that will be required, and the reliability allocated to this failure mode.

5. List the component mechanism(s) of failure stemming from the success-failure condition, causes, and interactions; give the probability equation; do the probabilistic analysis (or assess by one of the other acceptable methods);[a] show the principal distributions; report the assessed reliability.

6. Determine how the mechanisms relate to one another (e.g., dependently or independently); combine the individual assessments to find the probability of success (reliability) under the failure mode.

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a. There are four acceptable methods for assessment of reliability. Values may be obtained by Analytical Estimation, Direct Measurement, Historical Data or Engineering Judgment. These are listed in order of preference.

Analytical reliability estimations were made by probabilistic analysis and the stress-strength interference principle. Probabilistic analysis is the fundamental method used to include variability in conventional engineering analyses. The principles of probabilistic analysis have been presented in several papers, reports, and books; and the method is an expansion of the analytical engineering sciences. It is equally applicable to structures, heat transfer, fluid flow, dynamics, electronics, etc.

Review of the NERVA program leads to the philosophy that flight safety considerations should start with the design process and provide solutions that are built into the nuclear design, not added on later. This approach minimizes weight, cost, and risk. Its purpose is to systematically determine the effects of all possible failures, suggest countermeasures to prevent a nuclear accident, assess the cost and benefits of mitigation, and recommend appropriate remedies. A major objective is to reduce radiological risk to the biosphere to acceptable levels under both normal and accident conditions. In support of flight safety, an extensive experimental data base needs to be established in materials, components, subsystems, and systems testing.

2.6.2 Safety Policy, Guidelines, and Review Process

United States policy on the use of nuclear reactors in space has been presented in a number of papers to the United Nations Scientific and Technical Subcommittee on the Peaceful Uses of Outer Space[6,7] and in the U.S. concurrence to the reports issued by that subcommittee.[8,9,10] The U.S. position requires that stringent design and operational measures be used by the U.S. to minimize potential interaction of radioactive materials with the populace and the environment and to keep exposure levels within limits established by international standards.

The U.N. Working Group believes that the bases for a decision on a nuclear power source should be technical provided that exposure risk is maintained at an acceptably low level. The Working Group defines that level by recommending that the annual dose equivalent limit for workers be set at 50 mSv (5 rem) whole body dose (or equivalent doses to parts of the

body). Furthermore, an annual dose equivalent limit for the most highly exposed members of the public (the critical group) of 5 mSv (0.5 rem) from all man-made sources should not be exceeded during the normal phases of a nuclear power system mission. The Group has not yet set specific guidelines for accident conditions.

DOE Safety Criteria

U.S. safety guidelines are further delineated in DOE criteria[11] and the current space nuclear power program, SP-100, specifications. These safety criteria and specifications require that credible launch pad, ascent, abort, or reentry accidents resulting in Earth impact not result in a sustained nuclear fissioning source. Therefore, reactor material, whether scattered by an explosion or intact, must be well within national and international safety standards. The reactor is also required to have at least two independent systems to ensure shutdown. An orbital altitude boost system is to be provided by the mission agency (for short-lived orbit missions) to boost the reactor into high orbits for radioactivity decay following mission completion or upon mission failure. These policies are considered adequate under current circumstances.

Mission Safety Guidelines

Current mission operational guidelines are given in JSC 30307, "Nuclear Safety Guidelines for Space Applications," with a current update being proposed in BB00231. These guidelines aid in the elimination and/or control of nuclear-related hazards by addressing nuclear system design, nuclear support system design, operations during flight, and operations during ground activities. Hazards, defined as potential risks in a system, are categorized as collision, contamination, corrosion, electrical shock, explosion, fire, injury and illness, radiation exposure, and radiation and temperature extremes.

Ground personnel and general population limits are extracted from Title 10, Part 20, of the Code of Federal Regulations. A 4-km diameter, controlled exclusion area around the launch pad is called for during

prelaunch and launch activities. Launches containing radioactive materials are to be conducted with the prevailing winds away from populated areas. Special concerns are presented for systems with liquid metals; however, nuclear rockets do not contain any liquid metals. Provisions for detection and decontamination must be made at landing sites in the event a nuclear source is on-board and in the event of radiation leaks. Flight termination impact areas for nuclear hardware outside the continental shelf, preferably in deep ocean areas, are to be investigated to minimize hazards to the ecology and general populace. Safety and destruct systems such as Eurasian overfly are to be considered to reduce impact potential and release of radioactive material on the Eurasian continent. Radioactive payloads must be able to accomplish the following:

1. Withstand the worst-case pressure gradient associated with the most credible scenario for detonation of the liquid and/or solid rocket propellant on the launch pad.

2. Withstand the worst-case temperatures created by the most credible source of fire associated with the detonation and burning of the liquid and/or solid rocket propellant.

3. Withstand reentry from Earth orbit and impact on land or water with a reentry trajectory that will generate the highest credible mecnanical shock and vibration.

4. Withstand worst-case credible combinations of pressure gradients, temperature, and vibration associated with detonation of the launch vehicle at any time during the launch and ascent phase.

The following are also required:

1. A positive and permanent shutdown system for malfunctioning reactors and for reactors which have completed their missions.

2. A redundant, automatic means of reactor shutdown to control operation under all contingencies.

An important, proposed provision is that permanent disposal be in a solar orbit of at least 0.84 of the Earth's orbital radius.

Safety Review Process

Every United States nuclear-fueled power supply that is considered for use in space must undergo a safety review process. This process establishes that the potential risks associated with the nuclear energy source use are commensurate with the anticipated mission benefits. A formalized review process has been developed for evaluating the safety aspects of nuclear system launches. At the center of this process is the Interagency Nuclear Safety Review Panel (INSRP), comprised of representatives from the Department of Energy (DOE), the National Aeronautics and Space Administration (NASA), and the Department of Defense (DOD). These agencies are responsible for evaluating mission safety for each launch. DOD and NASA personnel are involved because these two government agencies have safety responsibilities and expertise, both as launching organizations and as use organizations of space nuclear power. DOE has statutory responsibility for the safety of space nuclear power systems.

The evaluation process consists of the following elements:

1. The lead or sponsoring agency directs the manufacturer of the nuclear power system (NPS) to write a preliminary safety analysis report (PSAR) or updated safety analysis report (USAR) describing all aspects of mission safety.

2. Safety analysis reports are distributed to the members of the INSRP and each member agency conducts its own review and critique of the PSAR or USAR.

3. A meeting of the INSRP is held with member agencies and their mission hardware contractors (launch vehicle, nuclear fuel, power system, space vehicle, etc.) in attendance. The results of the independent reviews are presented and discussed at this meeting.

Action items are generated to resolve any open questions or issues.

4. The power system contractor, with input from other agencies/contractors responsible for action items, writes an FSAR taking into account the FSAR critiques and any appropriate new information.

5. Elements 2 and 3 are repeated with the FSAR.

6. The INSRP generates a Safety Evaluation Report (SER) that accompanies the request for Presidential approval of the launch.

The SER is the risk assessment of the INSRP and is not simply a reissue of the FSAR of the nuclear power system developed. In addition to the information provided in the FSAR, the SER also contains analyses and tests performed by many technical people from government agencies, laboratories, and universities. The SER evaluates potential human exposures to radiation and the probability of exposure during all phases of the mission. The INSRP submits the SER to the heads of DOD, NASA, and DOE for their review with the INSRP recommendations/conclusions about the safety of the NPS. The key concept here is that the INSRP recommends and does not make any final decision. The head of the agency which wants to fly the NPS then must request launch approval from the President through the Office of Science and Technology (OST). The heads of the other two agencies represented on the INSRP may choose to support the user agency with statements of support. The OST will review the user agency requirements and may send the request to the National Security Council for review. The ultimate authority for launch and use of the NPS lies with the President of the United States.

Figure 30 shows the generalized sequence of events in this flight safety review process. Because safety features are designed into U.S. nuclear power sources from the very beginning, this safety review process is actually an integral part of the overall flight system development and in no way constrains the overall mission schedule.

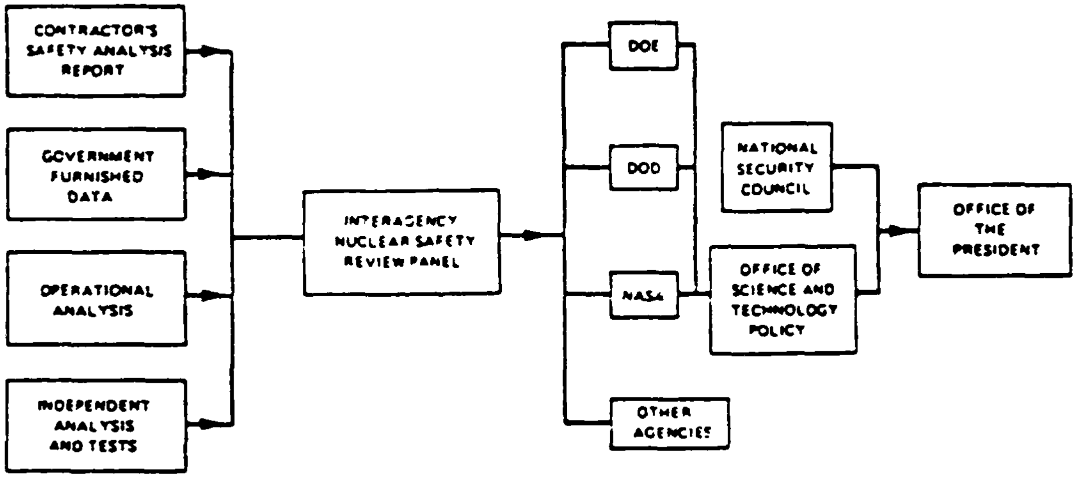


Figure 30. Safety review and launch approval process.

2.6.3 Operational Issues

Issues and approaches are addressed in terms of ground operations, launch, flight operations, disposal, and perceived safety. Test data and analysis from the NERVA program provide an extensive data base for future nuclear rocket programs.

Ground Operations

The principal issues concerning ground operations are (a) preventing accidental reactor criticality and (b) avoiding loss of special nuclear materials to terrorists. Approaches include shipment of the reactor in a special container (with the core heavily poisoned with neutron absorption materials in case of water immersion or compaction accident), use of a watertight structure, use of a shipping container that has been designed and tested for worst-case impact accidents, and shipment in a preferential manner. Such containers have been built, qualified (including sled impact testing into concrete walls), and used in the NERVA nuclear rocket program. Shipping containers have also been qualified for terrestrial reactor shipping vessels.

Launch pad operations safety considerations include not operating the reactor at a radiological level that requires restrictions on manned operations and providing independent and redundant neutron poisons in the reactor (including poisoned rods in the core channels).

Special handling issues relative to ground operations are (a) worker constraints in performing duties around a payload that includes a nuclear power plant and (b) the need for special handling equipment. The radiological levels in the vicinity of the reactor can be maintained well below established radiological standards by minimizing testing to zero power levels. A system of safety interlocks and mechanical key locks is also usually provided in the designs so that individual components can be tested prior to launch without permitting the reactor to go critical. The designs can also readily incorporate redundant and independent safety devices for worker protection.

Launch Safety

Launch is defined in this paper as the time from lift-off until the reactor is either inserted into Earth orbit or the reactor reaches a planned operational part of the flight plan.

The principal issues concerning launch safety are (a) preventing accidental criticality and (b) avoiding special nuclear materials being acquired by a foreign country. Redundant and independent neutron poisons can again be used to prevent inadvertent criticality. For NEP reactors, mechanical key locks can be used to avoid operation of the reactor control elements. Electronic locks can be used on direct thrust nuclear rockets. Suborbital flight operations, which apply to direct nuclear propulsion, can have a spectrum of failures such as (a) early failures yielding ballistic reentry and (b) late failures resulting in reentry from partial, low altitude, or highly eccentric orbits. The early failures would result in the nuclear power plant's impacting in deep water within the limits of the missile range. Here, the poison rods in the core can ensure subcriticality after water impact. Late nuclear stage propulsion or guidance failures could inject the nuclear propelled rocket into orbital flight paths of

minutes, days, or months. Reentry would be outside the missile range at a random impact point. Safety mechanisms include neutron poison insertion devices through the nozzle into the reactor core or additional scram rods in the core. Flight safety countermeasures can include reactor destruct devices activated to ensure that the debris lands in an ocean well away from land masses.

The loss of special nuclear material is preventable by selection of flight paths that minimize impact on land, and especially, on unfriendly territories. Destruction devices can be added to expendable launch vehicles to destroy the reactor over water.

Flight Operations Safety

The principal issues for flight operations concern (a) unplanned reentry into the biosphere, (b) restrictions on servicing reusable vehicles because of radiation buildup, and (c) radiological effects on satellites. Table 30 provides the lifetime of parking orbits. The shuttle can reach 100-200 nautical miles without the use of orbiting maneuverable systems (OMS) kits. Thus, the lifetimes are days to many months for corrective safety actions. A booster can be provided to increase the orbit in case of failure or at the end of its normal operational cycle. In addition, safety actions can take advantage of the space infrastructure to boost the reactor to higher orbits if a reactor failure occurs or at the end of its normal operational lifetime.

Acute guidance failures could yield low pedigree, elliptical short life orbits, or hyperbolic flight paths resulting in prompt random reentry. Safety interlocks can be used to prevent this. The angles of thrusting can be selected to more towards safer orbits. Engine destruct devices can also be used. In addition, anti-satellite weapons can be used to destroy "wayward" nuclear rockets.

Control systems for the reactor can be made redundant and logic elements can be used to determine deviations between the independent,

TABLE 30. PARKING ORBITS

Altitude (Nautical Miles)	Expected Time in Orbit (Days)
85	1/2
100	3
150	35
200	200
300	4000

redundant control systems. The logic elements would then reject failed elements.

Single stage to orbit has the same hazards to contend with as suborbital start plus the need to prevent radiological release into the atmosphere and the need for launch pad exclusion zones. Flight safety can be achieved by fuel element coatings.

Servicing can be performed using robotic vehicles. The procedures are similar to those for servicing chemical propulsion systems in space. Radiation hardening would be an additional requirement. Figure 29, using data supplied by Westinghouse Electric Corporation, shows the radiation intensity decay after engine operations from 2-20 min (typical orbital transfer times). Behind the shadow shield, the levels of radiation are low enough for limited, manned servicing without exceeding radiation limits of 5 rad/year. Limited, manned servicing is possible if a short period of time (days) is allowed for fission product decay. The radiological effects on the satellite are reduced to acceptable levels by configuration arrangements, positioning of liquid hydrogen tanks, and special radiation attenuation shielding materials.

Special space operational limits issues concern the need for special types of servicing equipment, radiation-hardened service, and the need for satellite equipment as well as limitations on manned operations. The major modification to a chemical depot in space would be the provision for radiation hardening the servicing equipment. Maintenance on the nuclear

rocket would likely be restricted to external components. Most servicing would be performed using robotic equipment whether chemical or nuclear tugs are involved.

Final Disposal

The principal safety issues in final disposal are long-term orbit contamination and random reentry into the biosphere. A suggested approach is to use the space infrastructure and attach booster rockets to move the spent reactors to a permanent disposal site.[12] Multiple boost attempts can be made, if necessary, until success is achieved. Operated space reactors should probably never be returned to Earth in order to minimize risk to the Earth's population.

2.6.4 Summary

Significant points concerning the safety of nuclear powered rockets are summarized below.

1. The use of nuclear reactors in space is accepted and provided for by U.S. and U.N. policies.

2. Safety specifications and criteria exist for reactors used in space.

3. Design practices exist, based on Rover/NERVA experience, for safely designing nuclear rockets.

4. A safety technology base supported by extensive experiments has been developed.

5. A safety approval process is in place.

Thus, high confidence exists that space nuclear power sources can be developed to meet the technical demands of design and operations criteria as well as the requirements for public and national safety.

3. TEST PROGRAM AND FACILITY REQUIREMENTS FOR QUALIFYING NUCLEAR ROCKET FUEL

The development of a nuclear rocket will require testing (both nuclear and non-nuclear) prior to final demonstration tests in an integral engine test. The subject of this section is to define the test program and facility requirements needed for qualification testing prior to integral engine testing.

The performance of a rocket is measured in terms of the specific impulse (I_{sp}) which is defined[13] as:

$$I_{sp} = \frac{\text{thrust}}{\text{mass flow rate of propellant}} = AC_f \sqrt{T_c/M} \quad ,$$

where

A	=	a performance factor related to the thermophysical properties of the propellant,
C_f	=	the thrust coefficient which is a function of the nozzle,
T_c	=	the chamber temperature, and
M	=	the molecular weight of the exhaust gases.

The rocket performance increases as the propellant gas temperature increases and as the molecular weight of the exhaust gases decreases. Consequently, hydrogen as the lowest molecular weight gas would normally be chosen for nuclear rockets. Potentially, nuclear rockets can provide about twice the specific impulse of the best chemical rockets. The use of a single propellant also results in logistical and safety advantages.

Because a nuclear rocket can operate over a relatively wide range of thrust output by varying the reactor power and propellant flow rate, it can perform a wide spectrum of space missions. Nuclear rocket propulsion would be useful for orbital transfer, fast launch space interception, and upper stage to orbit missions. The nuclear rocket would not be operated until after being launched with a chemical rocket. The initial operating goal

for a nuclear rocket engine is to provide ~10,000 to 30,000 lb thrust at high temperature (2700 to 3000 K) for a total of 10 hr, spread over ~160 reactor operating periods of 2 to 8 min each.

The two most prominent reactor concepts currently proposed for space propulsion are the NERVA-derivative reactor and the fixed particle bed reactor. Other reactor concepts being proposed for a nuclear rocket include the Los Alamos National Laboratory's Prismatic reactor (SPR-9) based on the Tory IIC ramjet reactor and General Electric Company's 710-derivative reactor with cermet fuel of UO_2 in tungsten. The NERVA-derivative reactor[14,15] concept, proposed by Westinghouse Electric Corp., is based on several NERVA solid fuel reactors that were developed and tested from 1955 to 1973 as part of the ROVER program. The particle bed reactor,[16] proposed by Brookhaven National Laboratory and Babcock and Wilcox Co., is based on the extensive experience attained in the development of coated-particle fuels for the High Temperature Gas Reactor program. The particle bed reactor concept has not been tested in a nuclear environment at high temperature.

This report presents test plans and facility requirements for qualifying fuel elements to be used in either of the nuclear rocket designs. Individual or a whole-core of fuel elements would be tested in a nuclear facility to determine maximum operating limits. Some of the fuel elements would be tested to failure to determine operating margins and fuel failure modes and consequences. A qualified fuel element design would then be tested in a full-scale nuclear engine facility for final qualification. Fuel failure would not be a planned event in the nuclear engine facility because of the problems and expense involved with large scale contamination cleanup and core replacement.

Section 3.1 describes each of the two nuclear rocket fuel concepts and their current development status. The anticipated failure modes and consequences of each concept are discussed in Section 3.2. Proposed test plans for qualifying each fuel concept in either a test loop-driver core or a small-reactor facility are given in Section 3.3 The general design specifications and operating requirements for the reduced scale nuclear

facility are described in Section 3.4. A summary of the report is given in Section 3.5.

3.1 Nuclear Rocket Concepts

The NERVA derivative reactor concept, proposed by Westinghouse Electric Corp., consists of an array of hexagonal fuel elements composed of UC and ZrC dispersed in a graphite matrix, (U,Zr)C-C. Hydrogen coolant flows axially through small-diameter channels. All surfaces of the fuel elements including the coolant channels are coated with ZrC to inhibit corrosion-erosion by the hot hydrogen gas. These fuel elements were demonstrated in the NERVA program for up to 1-hr continuous operation with hydrogen gas coolant temperatures of 2450 K.

The fixed particle bed reactor (PBR) concept, proposed by Brookhaven National Laboratory and Babcock and Wilcox Co., is based on the extensive experience gained from the High Temperature Gas Reactor program. The concept uses an array of fuel elements which are composed of carbon and ZrC coated UC_2 particles (0.5 mm diameter) that are contained between two porous cylindrical screens (termed frits). The fuel elements are surrounded by a solid moderator such as ZrH_2 or ^7LiH. Coolant flows axially through channels in the moderator, then radially inward through an outer frit, the fuel-particle bed, and then through an inner frit into a central channel where it exits into the exhaust nozzle. The concept has not been tested in a nuclear environment or at high temperature.

This section will describe each concept and its current development status.

3.1.1 NERVA Reactor Concept

Project ROVER was initiated in 1955 to develop a nuclear rocket engine to provide propulsion for a manned mission to Mars. The reactor concept selected (shown in Figure 31) was a solid-core, hydrogen-cooled reactor that expanded the exiting gas through a nozzle and discharged the gas into space. The objective of the NERVA nuclear rocket engine development

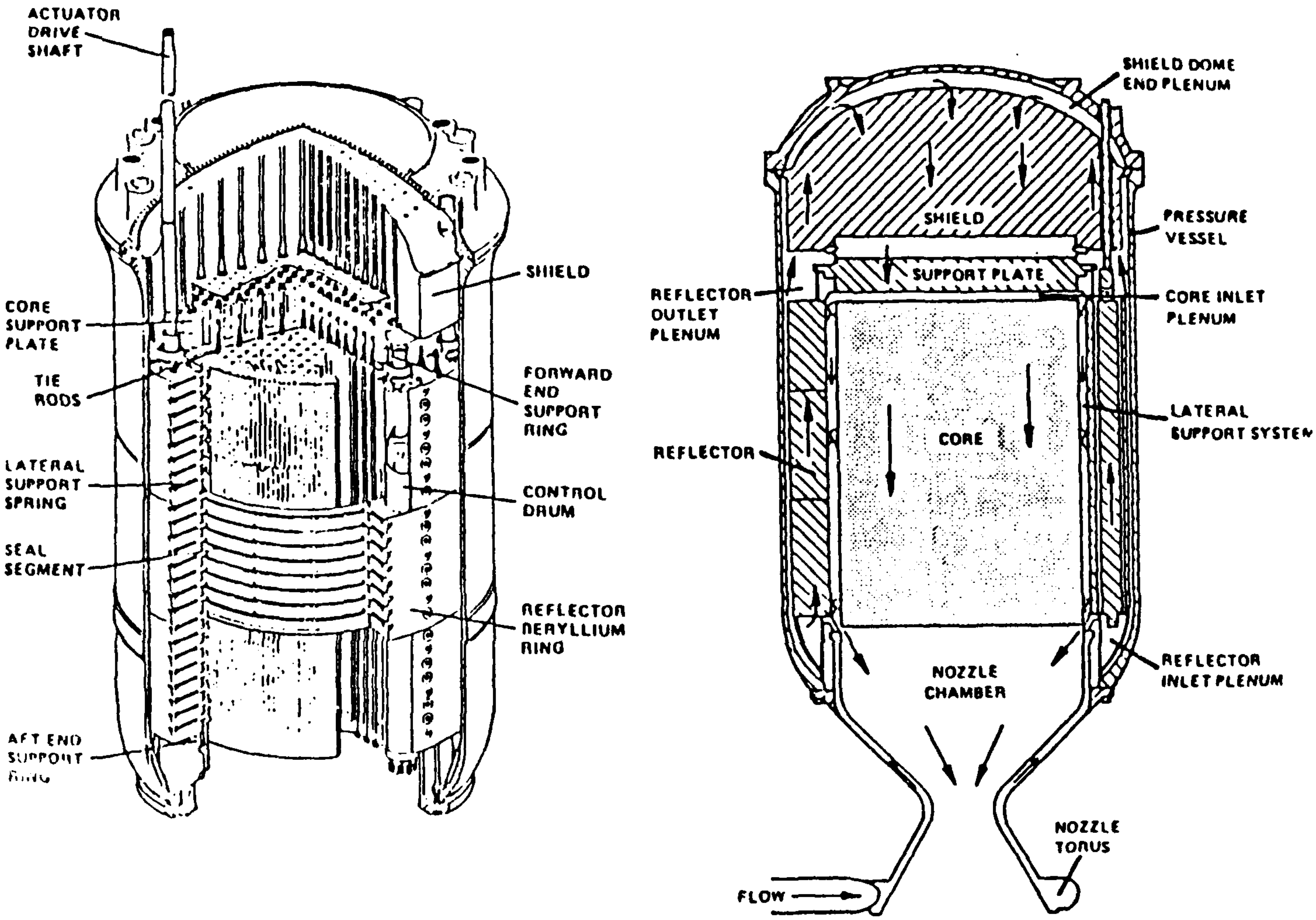


Figure 31. Schematic of NERVA engine.

program was to achieve the highest possible propellant temperature for the launch from earth and several hours of total operating time to travel to Mars and return. This goal implied that a very strong technology development program in reactor fuels would be required.

The Los Alamos Scientific Laboratory [now called the Los Alamos National Laboratory, (LANL)] was given the role of establishing a basic reactor design and of leading the fuel development effort.[13,17] Other key investigators were the Aerojet General Corporation for rocket engine development and Westinghouse Electric Corporation for nuclear reactor development. It was at this time that the acronym, NERVA, was established for the Nuclear Engine for Rocket Vehicle Application.

Only a few materials, including the refractory metals and graphite, are suitable for use in reactors designed to operate at very high temperatures. Figure 32 shows the metallurgical temperature reference points of interest. The metals are all strong neutron absorbers, whereas graphite is not. In fact, graphite, in addition to having excellent high-temperature strength, also acts as a neutron moderator and minimizes the amount of enriched uranium required in the reactor core. One great disadvantage of graphite, however, is that it reacts with hot hydrogen to form gaseous hydrocarbons and, unless protected, quickly erodes away. Consequently, one of the greatest challenges in the nuclear rocket program was to develop fuel elements of adequate lifetime in a high-pressure, hot hydrogen environment.

Temperatures above 2773 K were considered necessary for fuel elements and exhaust gases, but little information on the behavior and compatibility of materials and fuels at very high temperatures was available. Consequently, much work had to be done to gain a complete understanding of the behavior of materials in a nuclear rocket engine.

The first ROVER reactor (KIWI-A) was designed and built to produce $\sim$100 MW. The design involved uranium-loaded graphite fuel plates. The uranium loading of each plate differed to provide a relatively flat radial fission distribution.

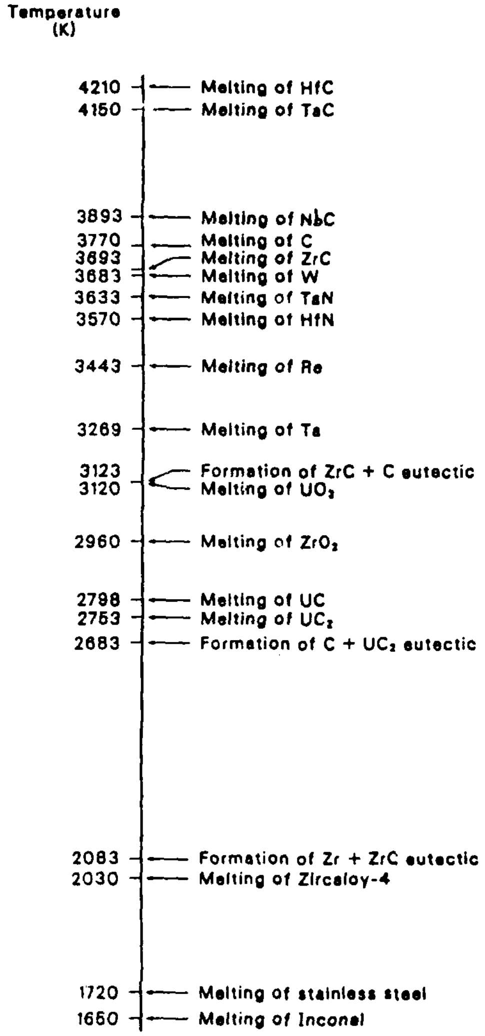


Figure 32. Metallurgical reference temperature points.

KIWI-A was tested at Nevada in 1959, for 300 s at 70 MW using gaseous hydrogen at 3.2 kg/s as the propellant. The fuel was hot enough to melt the UC_2 fuel particles (the UC_2-carbon eutectic temperature is 2683 K). The KIWI-A fuel plates were not clad or coated. Only the one reactor with plate-type fuel elements was produced and the technology was not representative of that developed for the nuclear rocket engine fuel elements.

KIWI-A prime (tested about 1 year later) was designed to test a new core configuration. Instead of fuel plates, it had 19-mm diameter fuel

elements, each containing four axial holes for propellant flow. Stacked fuel elements were contained in ∿1.372 m long, high-density, commercial graphite modules. The individual elements consisted of 4 μm diameter particles of highly enriched uranium dioxide (UO_2) extruded in a carbide matrix. The fuel elements were coated with niobium carbide (NbC) by the chemical vapor deposition (CVD) technique to protect against hydrogen corrosion. Some core structural damage occurred during a ∿6 min operation at 85 MW.

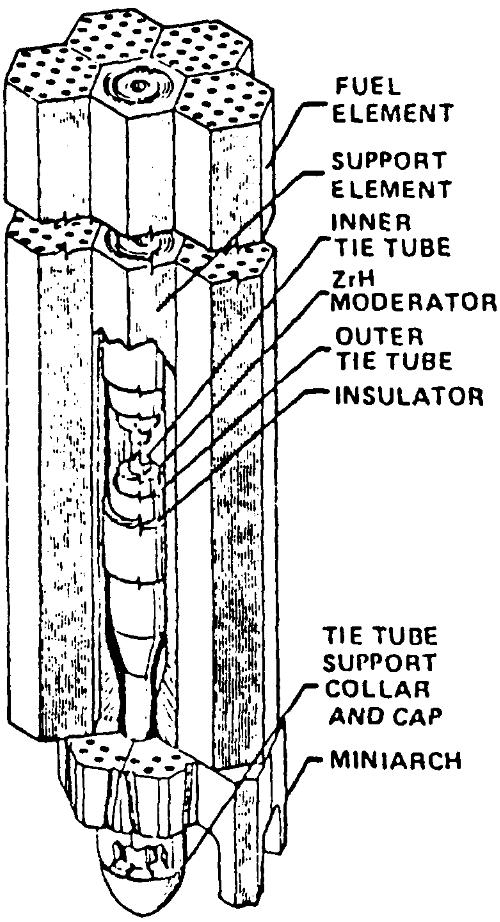
The experience gained with coating techniques for the KIWI-A prime reactor suggested modifications of the coating for the fuel used in the KIWI-A3. Basically, the CVD temperature was increased, which improved the coating adherence and provided thicker NbC coating at the same time.

The KIWI-A3 reactor was operated for ∿5 mins at 100 MW. Although core structural damage occurred, the general appearance of all fuel elements was excellent. There were several elements showing blistering and corrosion, but not sufficient to damage the modules.

The elements for KIWI-A through KIWI-B4D had UO_2 original fuel loading instead of UC_2. During fabrication the UO_2 was converted to UC_2, with evolution of CO and loss of carbon from the element. Temperatures could go as high as the UC_2-C eutectic temperature, 2683 K at which point the fuel melted. The major problem with UO_2-loaded fuel elements was that micro-size UC_2 particles, being extremely reactive, revert to UO_2 in the presence of air, particularly humid air. Thus, oxide-carbide-oxide reactions occurred during each heating and storage cycle, including graphitizing, coating, and reactor operation, and each cycle caused loss of carbon by CO gas evolution and degraded the element. The solution to the problem was the use of 50- to 150 μm diameter UC_2 particles coated with ∿25 μm of pyrolytic graphite. They were introduced with the KIWI-B4E reactors and used from 1964 to 1969 in all LANL and Westinghouse Astronuclear fuel elements.

From the design of KIWI-B1A with 0.66-m-long cylindrical fuel elements with 7 coolant holes, the design changed for KIWI-B4 to 1.32-m-long

hexagonal fuel elements with a flat-to-flat dimension of 19.1 mm and 19 coolant holes. This became the basic graphite fuel element design throughout the remaining Rover program, and is shown in Figure 33. The NERVA design in all of the Westinghouse NRX-A reactors used the same basic fuel element as the Rover reactors.



FUEL

- FUNCTION
  - PROVIDED ENERGY FOR HEATING HYDROGEN PROPELLANT
  - PROVIDED HEAT TRANSFER SURFACE
- DESCRIPTION
  - ^{235}U IN A COMPOSITE MATRIX OF UC-ZrC SOLID SOLUTION AND C
  - CHANNELS COATED WITH ZrC TO PROTECT AGAINST H_2 REACTIONS

TIE TUBES

- FUNCTION
  - TRANSMIT CORE AXIAL PRESSURE LOAD FROM THE HOT END OF THE FUEL ELEMENTS TO THE CORE SUPPORT PLATE
  - ENERGY SOURCE FOR TURBOPUMP
  - CONTAIN AND COOL ZrC MODERATOR SLEEVES
- DESCRIPTION
  - COUNTER FLOW HEAT EXCHANGER OF INCONEL 718
  - ZrH MODERATOR
  - ZrC INSULATION SLEEVES

Figure 33. Schematic of NERVA fuel element design.

The KIWI-B4E, Phoebus, Pewee, and NRX-A reactor tests used UC_2 particles with pyrolytic graphite coatings as protection against oxidation and storage, not for fission product retention. The particles were 50 to 150 μm diameter with a 25 μm thick coating. The fuel elements used a graphite matrix with NbC coatings to protect against hydrogen corrosion.

In later reactor designs, the NbC was changed to ZrC. This fuel demonstrated 1-hr operation at temperatures between 2400 and 2600 K. The major problems encountered with these designs involved the large difference between the coefficients of thermal expansion (CTE) of the graphite matrix and the coatings, resulting in cracking or loss of the coating. Excessive carbon loss after 1-hr occurred in the 2375 to 2575 K temperature range.

Development of new fuel elements that would be highly resistant to hydrogen attack and withstand high operating temperatures was undertaken. Three approaches were considered.

1. Changing the graphite flour and binder to new materials whose CTE values matched that of the coating and using only graphitizable constituents.

2. Retaining the 19-hole hexagonal design, but using a composite carbide and graphite matrix.

3. Designing a new all-carbide fuel element.

Much effort went into fabrication and evaluation of high-CTE graphite and composite carbide and graphite elements, and a lesser effort was devoted to the pure carbide fuel elements and other components during the development of high-temperature, long-life nuclear rocket propulsion reactors.

The final nuclear testing of developmental fuels[18] for the NERVA program was conducted in the Nuclear Furnace-1 (NF-1) reactor in 1972. The Nuclear Furnace was a heterogeneous water-moderated, beryllium-reflected nuclear reactor (see Figure 34) for testing fuel elements and other components of high-temperature, long-life nuclear rocket propulsion reactors. The full-power NF-1 operating conditions[7] are presented in Table 31. Of the reactor core's 49 fuel cells, 47 contained composite fuel elements (Figure 35) that varied in carbide content (30 or 35 vol%), coefficient of thermal expansion (6.1 or 6.7 μm/m$\cdot$K), and thermal-stress-resistance (4700 or 6200 MW/m^3 fracture power density).

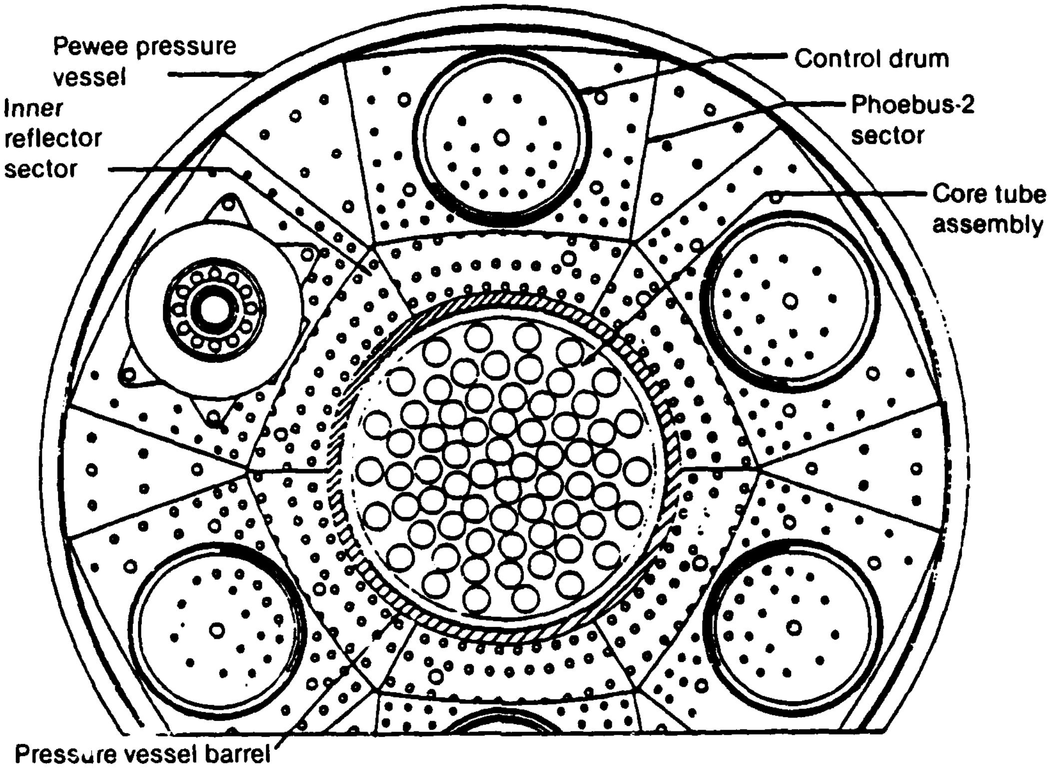


Figure 34. Cross-section of Nuclear Furnace - 1 core.

TABLE 31. NF-1 REACTOR FULL-POWER CONDITIONS

Parameter	Value
Reactor thermal power, MW	44
Power/composite element, MW	~0.90
Hydrogen flow/element, kg/s	0.022
Hydrogen-inlet pressure, MPa	4.7
outlet pressure, MPa	3.2
inlet temperature, K	340
outlet temperature, K	2450
Maximum matrix temperature, K	~2500
Peak power density, MW/m^3	~4500
Run time at full power, min	109

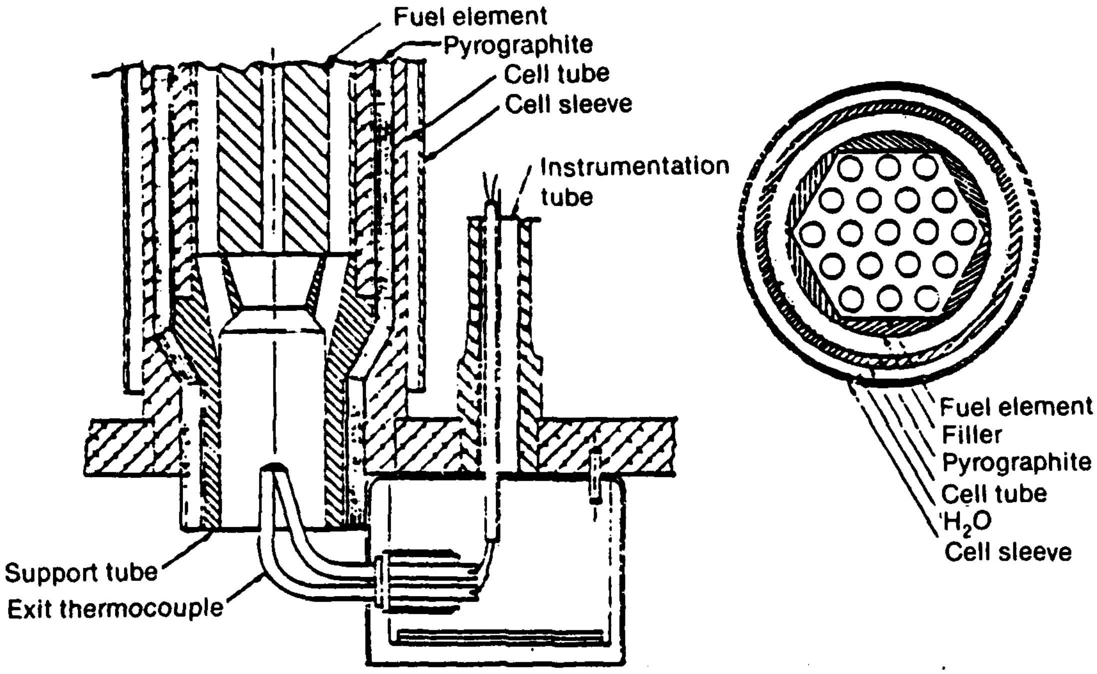


Figure 35. Axial view of lower region of fuel element and cell and cross-section of fuel element and cell of Nuclear Furnace - 1.

The experimental composite elements tested in NF-1 confirmed the belief that minimizing the thermal expansion mismatch between the coating and fuel element matrix would reduce coating cracks and carbon mass loss. The composite fuel elements withstood peak power densities of 4500-5000 MW/m^3 and outlet hydrogen gas temperatures of 2450 K without major difficulties. The usefulness of composite elements was limited by their apparent susceptibility to radiation damage of the ZrC coating.

Two of the 49 cells in the NF-1 reactor contained hexagonal (U,Zr)C solid-solution, all carbide elements, 5.5 mm across the flats, 0.66 m long, with a single coolant hole in the center. Each cell contained seven elements, bundled together, as shown in Figure 36, with another seven elements stacked on top to complete the 1.32-m length of the cell, for a total of 14 elements per cell.

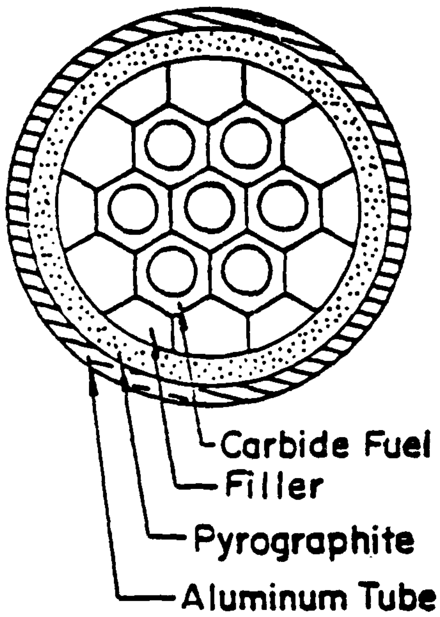


Figure 36. NF-1 reactor cell containing (U, Zr) C(carbide) fuel elements. The filler was ZrC-graphite (composite) coated with ZrC.

Carbides have poor thermal stress resistance; however, even if the fuel element cracked during the reactor run there would be no carbon loss problem. The major area of concern was possible crumbling of the carbide fuel element into particles that would block passage of the hydrogen gas. The primary purpose for testing the carbide fuel elements was to determine the fracture mode at high ($\sim$4500 MW/m^3) power densities. The 28 experimental carbide fuel elements tested in NF-1 were impregnated with 0, 3, or 8% zirconium. The test conditions caused many transverse and some longitudinal fractures, but no fragmentation into small particles. The 8% Zr-impregnated carbide elements were less fractured than the other two types.

Summary of NERVA Concept

Development of fuel elements for nuclear rocket engines presented a unique challenge to materials development workers. Reactor fuel elements had never been operated at temperatures up to 2773 K in a hydrogen environment, and the technical problems were, indeed, formidable.

Early NERVA test results indicated that there were major problems related to the loss of graphite element material to the hydrogen coolant. Two mechanisms by which this loss occurred were identified:

1. Diffusion through the coolant channel coating and cracking of the coating. The coating (either NbC or ZrC) was applied to the coolant channels of the fuel elements using a CVD technique. The purpose of the coating was to prevent corrosive interaction between the graphite and the hydrogen coolant, which would lead to loss of graphite material and subsequent loss of reactivity, and even element integrity. Diffusion of graphite through the coating occurred predominantly at the gas exit end of the reactor (termed the "hot end") where the coolant temperatures were highest. For coolant temperatures below about 2000 K, diffusion of graphite through the coating was not a problem. Several changes in the element coating process were made that essentially eliminated graphite diffusion through the ZrC. These changes included minor increases of the coating thickness, improved deposition techniques, and an increase in the temperature at which the coating was applied.

2. The more serious graphite loss problem involved the second mechanism, in which graphite diffused through cracks in the coating. This problem occurred predominantly in a region about one-third of the core length away from the core entrance. At this location, termed the midrange region, graphite diffusion directly through the coating was not a problem because of the lower coolant temperatures. However, the lower coolant temperature, in conjunction with the higher local power density in this region, resulted in the most severe temperature gradient across the fuel element. This radial temperature gradient gave rise to stresses that were sufficient to crack the coating, thereby allowing graphite diffusion to the coolant and/or hydrogen reacting with the graphite. Cracks were also formed by radiation damage, apparently due to interaction of fission fragments with the graphite, which degraded the thermal conductivity and caused high temperature gradients and cracking

of the coating. Damage to the coating also occurred during the CVD process itself as the coating cooled from its deposition temperature.

As the NERVA tests proceeded, changes in the coating procedure were made that steadily reduced the cracking problem. These changes included thinner coating in the midrange region, variation of CVD temperature to control crack size and to improve adhesion, the use of a molybdenum overcoat to fill in microcracks formed during cooling of the applied coating, radial power flattening via orificing and enrichment zoning to reduce thermal stresses, and producing graphite with a coefficient of thermal expansion that matched that of the coating. By the end of the NERVA development program, the rate of graphite loss from the elements had been reduced by a factor of 10. For operation at full power for about 110 min, tests in NF-1 resulted in graphite mass loss of about 1% to 2% compared to 10% to 20% for earlier tests. In addition, although the extent of cracking in the carbide elements tested in the NF-1 was disappointing, the design and development effort for carbide elements had been minimal and it was felt that much could be done to improve behavior and minimize the thermal stresses during reactor operation. The very high temperature capability of all these carbide elements in a hydrogen environment provides a unique capability for future reactor concepts.

An orbital transfer vehicle nuclear rocket using NERVA-derivative fuel elements with a total U content of R25 kg to produce a power of 200 MW (thrust of 10,000 lb) for a total operation of 10 hr would involve a fuel burnup of R3300 MWd/MtU.

Although the NERVA program was terminated some 15 years ago, the technology was well documented and is considered to be approximately 90% retrievable. After the NERVA program ended, Westinghouse continued to refine the fuel and coating materials to the point that graphite loss was determined to be insignificant during electrical heating tests.

3.1.2 Particle Bed Reactor Concept

Although a particle bed reactor has not been constructed yet, several designs of particle bed reactors have been proposed for nuclear rocket use. Three basic design variations have been proposed: the fixed particle bed reactor, the rotating bed reactor, and the multiple-fuel element particle bed reactor.

The fixed particle bed reactor consists of fuel particles configured in a single large annulus between porous screens (frits) with moderator and reflector material on the outer periphery and a graphite reflector-moderator plug located at the center of the reactor. Hydrogen gas enters through the outer frit, is heated as it passes through the particle bed, enters the inner frit, through a central channel formed by the inner frit, and exits through the nozzle.

The rotating bed reactor is composed of a fluidized bed of fuel particles that are held in place by centrifugal force inside a rotating cylindrical outer frit. Hydrogen gas enters through the rotating frit, is heated as it passes through the particle bed, enters a central cylindrical cavity, and exits through the exhaust nozzle. The degree of particle fluidization is controlled by the rotation speed. Presumably, the outlet gas temperature and the power density could be much higher than other reactor types because an inner frit may not be needed (except possibly for a "loss-of-rotation" accident). It is obvious that an extensive research and development effort would be required for this design. Both of these designs are relatively massive because of the large amounts of moderator and reflector material needed.

The latest proposal by Brookhaven National Laboratory and Babcock and Wilcox Co. for a particle bed reactor is the multiple fuel element design. This design, shown in Figure 37, consists of a hexagonal array of 19 fuel elements interspersed in a moderating material, which is surrounded by additional moderator and reflector material. As shown in Figure 38, the fuel elements consist of an annulus of coated fuel particles that are fixed between two porous cylindrical frits. The individual

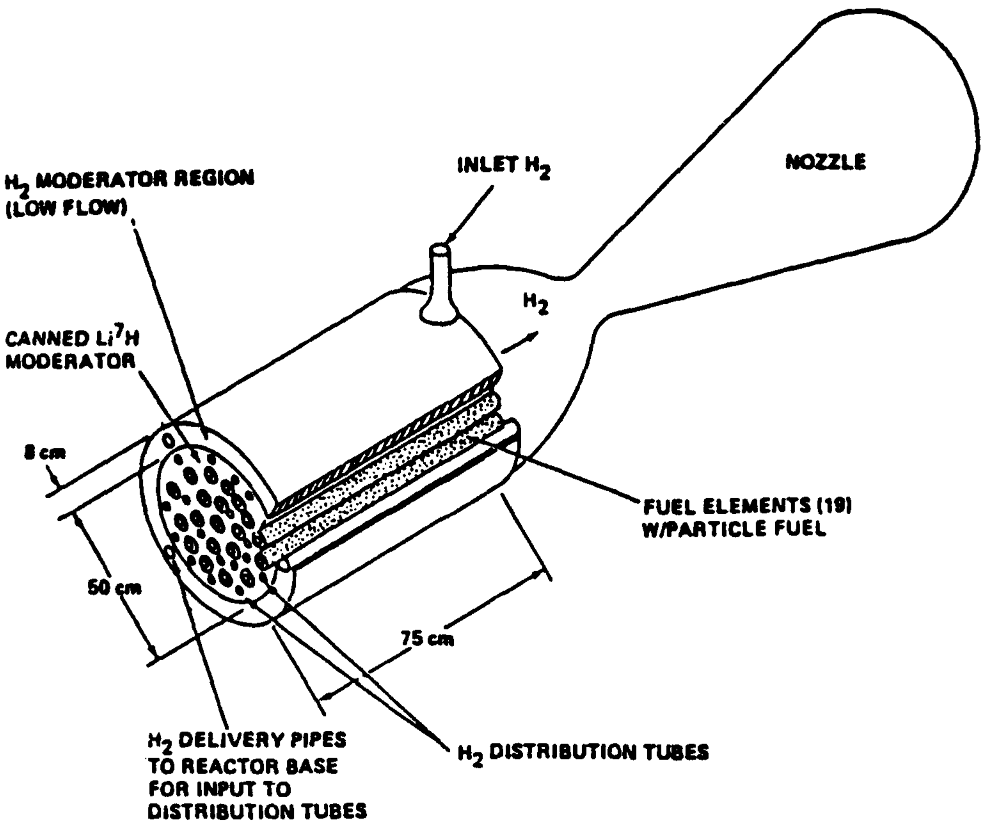


Figure 37. Schematic of particle bed reactor using high temperature graphite and ZrC coated fuel particles.

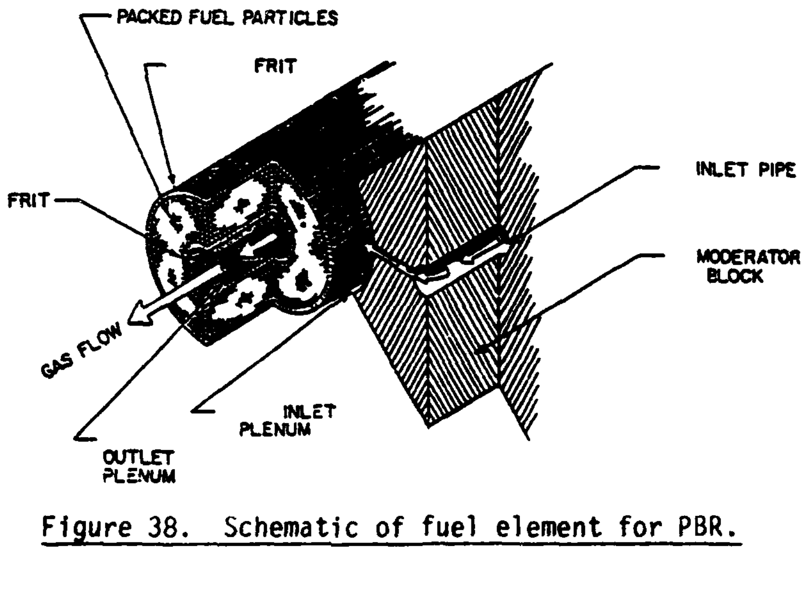


Figure 38. Schematic of fuel element for PBR.

0.5-mm-diameter fuel particles, shown in Figure 39, are composed of a uranium carbide kernel, a layer of porous graphite, a layer of pyrolytic graphite, and an outer layer of ZrC. The coolant flow system, shown in Figure 40, consists of multiple parallel flows connected to common inlet and outlet headers. The hydrogen coolant enters the reactor core from the bottom and is distributed to each fuel element through numerous inlet channels along the length of an Inconel outer frit. The hydrogen flows through the low-temperature outer frit, then through the packed bed of fuel particles, through the high-temperature ZrC inner frit, and then along the inner diameter of the hot frit until exiting through the exhaust nozzle. A schematic of the holes that produce a porosity of 50% in the frit material is shown in Figure 41. The active fuel length is 0.508 m, with 0.076 m length of graphite serving as an axial reflector above and below the active fuel region.

The moderator elements would be canned to prevent hydrogen loss. Small spacings between moderator elements would permit sufficient hydrogen flow for cooling the moderator material.

Reactivity control would be by rotating control drums placed in the reflector region. The drums would be of the same material as the reflector with a 120° sector covered by an absorbing material. The following materials are being considered for a PBR nuclear rocket:[20]

Fuel Particle Material. It is anticipated that the fuel kernel would have to be made of UC-ZrC to withstand fuel temperatures in excess of ~2600 K. UC_2 fuel kernels are presently planned for lower temperature applications (less than 2600 K).

Fuel coating material. The fuel particle could be coated with ZrC (melting point of 3693 K) or HfC (mp of 4210 K). HfC has a thermal neutron cross section of ~100 b so its use would have to be restricted to only the hottest part of the particle beds.

Hot frit material. The hot frit material would be chosen for its ability to withstand high temperatures, radiation damage, and thermal cycling.

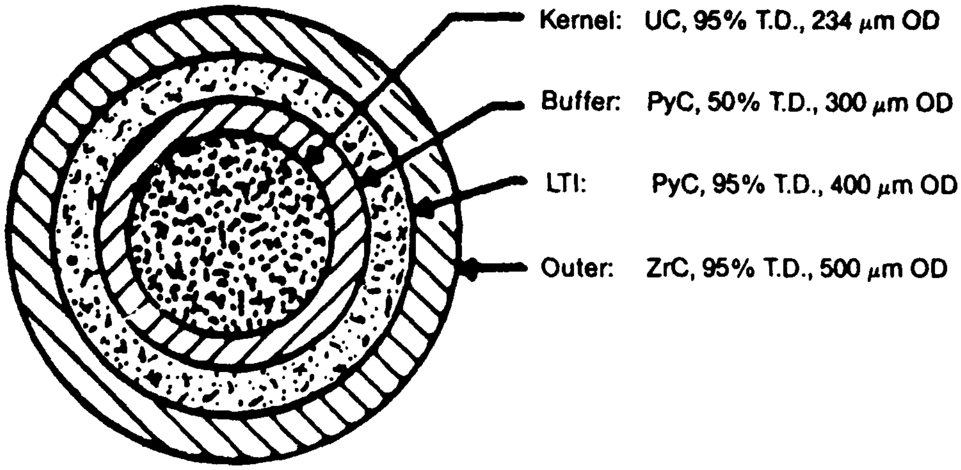


Figure 39. Schematic of fuel particle for PBR.

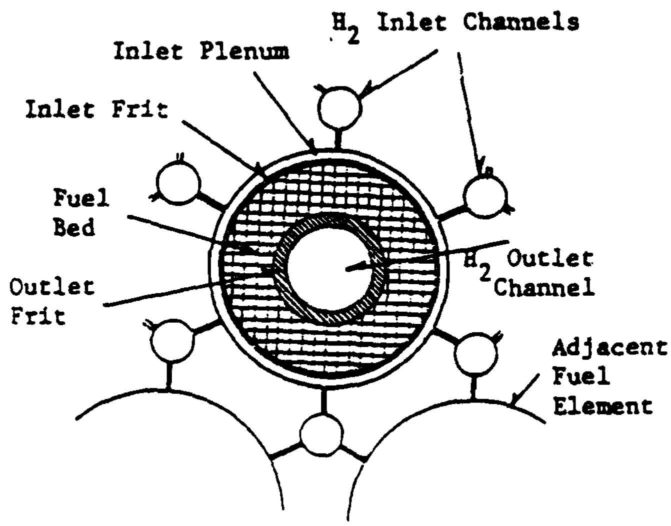


Figure 40. Schematic of hydrogen flow path for PBR fuel element.

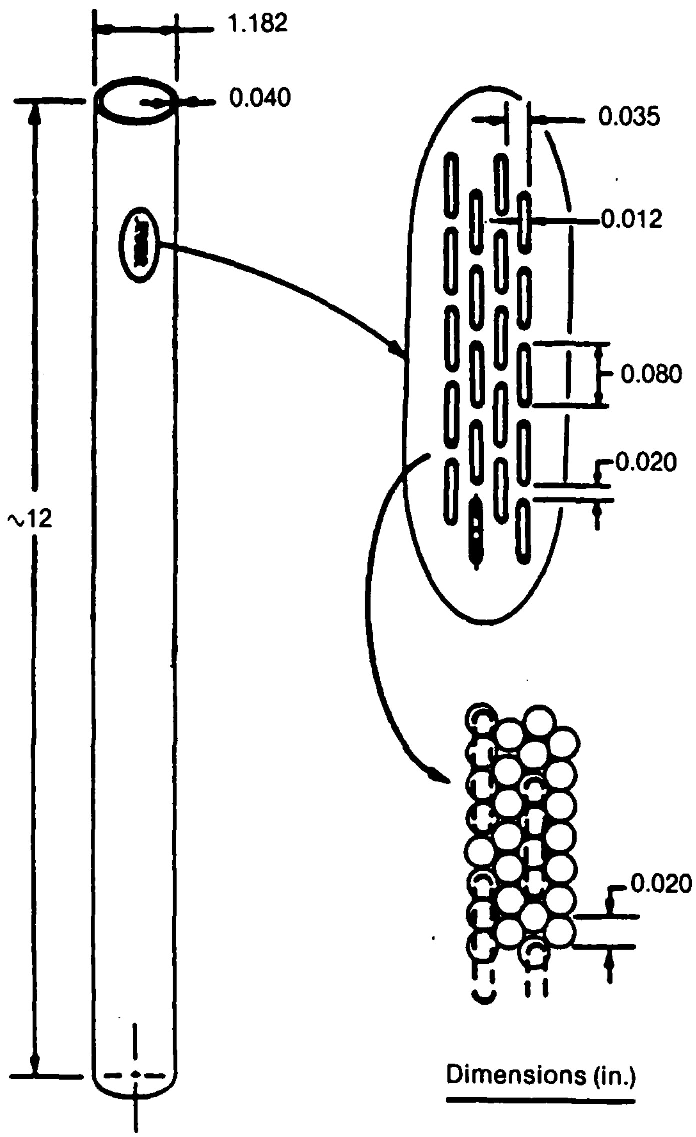


Figure 41. Schematic of outlet holes in hot frit of PBR fuel element.

Re (m.p. of 3443 K), ZrC (ZrC-graphite eutectic forms at ~3123 K), ZrC coated carbon, HfC coated carbon, or carbon are possible candidates. Re and HfC-carbon have relatively high thermal neutron cross-sections of 88 and 102 b and also are high resonance absorbers. Tungsten and a combination of W-Re have been tested, but were not suitable for use as a hot frit material.

Cold frit material. Inconel and stainless steel have been proposed for this low temperature location.

Reflector and Moderator Material. Zirconium hydride, lithium-7 hydride, and beryllium are possible candidates.

Other characteristics of the particle bed reactor are as follows:

Power Density. The reactor power and particle bed power density requirements determine the bed thickness and coolant flow rate. The flow velocity in the outlet channel is kept below a Mach number of 0.3 to minimize vibrations. A power density of 8.2 kW/cm^3 is currently proposed for a 200 MW orbital transfer vehicle (as compared to 3.6 kW/cm^3 for a similar NERVA OTV).

Coolant Pressure Drop. According to calculations, most of the coolant pressure drop in a particle bed reactor occurs as dynamic pressure head and losses at the outlet channel. Pressure drop is only slightly dependent on power density and axial power shape, but is strongly dependent on the outlet flow Mach number. For Mach numbers less than 0.3, the core pressure drop can be kept less than 0.4 MPa at an operating pressure of 7 MPa.

Development Status. The multiple fuel element particle bed reactor concept evolved in 1981 by Brookhaven National Laboratory (BNL) researchers from the rotating particle bed reactor concept, which was first proposed in 1960. The particle bed reactor concept retained the advantages of large heat transfer, high power densities, and small size, while it eliminated the problems associated with rotating the particle bed. Only a modest effort has been expended on developing this concept. Limited testing of

material has been done by BNL and by Babcock and Wilcox Co. Present testing activities have been directed towards the proposed use of the PBR for electric power generation in space. BNL has conducted two series of thermal tests on the PBR carbon coated fuel particles. In one test series, an electric heating coil was placed in the particle bed and the test assembly (consisting of depleted fuel particles packed between two frits) was heated to about 1500 K. In the other test, the frits were used as the anode and cathode of a direct heating circuit with electric current passing directly via the fuel particles to achieve a peak temperature of $\sim$2300 K. Hydrogen or helium gas flow was maintained though the assembly in both of these test series. Particles have also been heated in an electric induction furnace at BNL.

Non-nuclear and nuclear heating tests are planned to be completed next fiscal year for sample PBR fuel elements. The nuclear test,[21] designated as PIPE (Pulsed Irradiation of a Particle Bed Fuel Element), will be conducted at the Annular Core Research Reactor (ACRR) at Sandia National Laboratory. This test will provide the first fission-heated demonstration of a PBR fuel element. Only transient heating up to $\sim$2000 K for a few seconds will be possible, however, so additional steady-state high-temperature testing will be required.

There have been several proposals and papers presented for using the multiple-fuel element PBR concept in a nuclear rocket, but no high-temperature (>2500 K) testing has been performed to date. The most recent nuclear rocket proposal[16] uses 19 PBR fuel elements with a total UC content of 10 kg to produce a power of 200 MW (thrust of $\sim$10,000 lb), a hydrogen flow rate of 5 kg/s, with an outlet gas temperature of 2750 K. The dimensions of the fuel assemblies proposed are: 0.508 m active length, 0.73 m overall length including graphite reflector pieces, 1.9-cm inner diameter and 2.5-cm outer diameter of the hot frit, and 6.2-cm inner diameter and 6.5 cm outer diameter of the cold frit. A fuel burnup of about 9000 MWd/MtU would be produced during a total of 10 hr operation at 200 MW. For the remainder of this section it will be assumed that this preliminary PBR design will be approximately correct regarding dimensions and flow rate.

3.2 Anticipated Failure Modes

This section lists the potential failure modes and effects that were foreseen by a nine-member panel of nuclear fuel experts for the NERVA-derivative and particle bed reactor concepts. The probability of a particular failure mode occurring was not considered in this analysis. A range of possible consequences is given for each failure mode since the severity and extent of damage would be expected to vary.

3.2.1 NERVA-Derivative Concept Failure Modes

The NERVA-derivative potential failure modes are listed in Table 32 according to initiating event, possible consequences after failure occurs, and optimum facility for investigating the failure mode. The potential failure modes that are closely related are grouped together.

Most of the failure modes are related to cracking of the ZrC coating on the outer surface of the fuel elements. The fuel failure modes listed here are primarily for operation above 2450 K since an extensive fuel development and testing program up to $\sim$2450 K was completed during the NERVA program.

3.2.2 Particle Bed Concept Failure Modes

No nuclear testing and very little electrical heating testing has been performed to data for the particle bed concept. Therefore, the extent of operating problems to be encountered and the development effort that will be required to achieve high temperatures for 160 cycles with a total of 10 operating hours is relatively unknown. The potential failure modes that were foreseen by a nine-member panel are listed in Table 33, along with possible failure consequences, and the optimum facility for investigation. Most of the potential failure modes are related to failure of the fuel particle coating layers of C and ZrC, subsequent blocking of the hot frit by the coating debris, and overheating. If the coating layers fragment instead into small pieces that are swept through the holes in the hot frit

TABLE 32. NERVA--DERIVATIVE POTENTIAL FAILURE MODES

Failure Mode Initiating Event	Possible Anticipated Consequences	Optimum Facility For Investigation
1. Fuel element coating of ZrC cracks or comes loose (may be radiation damage)	Material loss from reactor with hydrogen, enhanced fission product release, reduced operating time	Small-scale nuclear tests as function of temperature and burn-up, post-test hot cell examinations (PIE)
2. Erosion/corrosion of material by hydrogen (also includes hydrogen embrittlement of reactor components	Material loss from reaction with hydrogen, enhanced fission product release, reduced operating time, non-critical geometry	Small-scale nuclear tests as function of temperature, burn-up, and hydrogen flow rate, PIE
3. Fatigue and thermal stress-strain problems (may be burnup dependent)	Fracture of graphite elements, loss of ZrC coating integrity, material loss, enhanced fission product release, reduced operating time.	Small-scale tests as function of temperature, thermal cycling, burn-up, and temperature ramp rate, PIE
4. Large radial or axial temperature gradient	Fracture of graphite element or ZrC coating. Material loss, enhanced fission product release, reduced operating time.	Small-scale nuclear test as function of temperature, PIE
5. High temperature	Melting of UC-ZrC-C fuel material, enhanced fission product release, loss of material, melting of adjacent structures, reduced hydrogen flow, reduced operating time.	Small-scale nuclear tests as function of temperature and time at temperature, PIE
6. Enhanced fission product release from fission gas bubble formation and high temperature	Shattering of fuel matrix, reaction of fission products with graphite or ZrC, loss of geometry, fuel swelling, reduced hydrogen flow, high temperature melting, reduced operating time.	Small-scale nuclear testing as function of temperature and burnup, PIE
7. Thermal expansion from 4 K to ∿3000 K	Differential expansion causes structural failure, loss-of-flow, material loss, reduced operating time.	Non-nuclear tests, integral engine test.
8a. Flow control valve or turbo pump malfunction, reduced hydrogen flow, no hydrogen flow	ZrC coating failure, meltdown of fuel, non-operable reactor, enhanced fission product release, reduced operating power and temperature.	Small-scale nuclear tests as a function of flowrate and temperature, PIE
8b. Flow reduction due to material relocation blocking coolant channels	ZrC coating failure, meltdown of fuel, non-operable reactor, enhanced fission product release, reduced operating power and temperature.	Small-scale nuclear tests as a function of flowrate and temperature, PIE

TABLE 32. (continued)

	Failure Mode Initiating Event	Possible Anticipated Consequences	Optimum Facility For Investigation
8c.	Reduced flow for decay heat removal	ZrC coating failure, meltdown of fuel, non-operable reactor, enhanced fission product release, reduced operating power and temperature.	Small-scale nuclear tests as a function of flowrate and temperature, PIE
9.	Power transient caused by liquid hydrogen, control system malfunction, or improper startup procedure	Failure of ZrC coating, fuel meltdown, structural failure, loss of material, reduced operating time, enhanced fission product release	Special transient nuclear facility, PIE
10.	Not able to shut down or start up reactor because of control element or control system malfunction	Loss of maneuverability, fuel failure, deplete hydrogen supply	Non-nuclear testing
11.	Radiation or thermal stress effects on moderator-reflector material	Loss of reactivity, loss of material, structural failure of fuel support components, reduced operating time	Non-nuclear tests, integral engine tests, PIE
12.	Vibration of reactor and engine components	Structural failure of fuel support components, loss of material, reduced operating time	Non-nuclear tests, integral engine tests, PIE

TABLE 33. PARTICLE BED CONCEPT POTENTIAL FAILURE MODES

Failure Mode Initiating Event	Possible Anticipated Consequences	Optimum Facility For Investigation
1a. Failure of fuel particle coatings (C and ZrC) from thermal cycling or fission product reaction or pressure	Outlet frit blockage, high fuel temperatures, frit failure, enhanced fission product release, reduced operating time, loss of particles	Small-scale nuclear tests as function of temperature and burnup, PIE
1b. Erosion-corrosion of particle coatings from hydrogen flow	Outlet frit blockage, high fuel temperatures, frit failure, enhanced fission product release, reduced operating time, loss of particles	Small-scale nuclear tests as function of temperature, PIE
1c. Erosion of particle coatings from vibration against frits or each other	Outlet frit blockage, high fuel temperatures, frit failure, enhanced fission product release, reduced operating time, loss of particles	Small-scale nuclear tests as function of temperature, PIE
1d. Manufacturing defects	Outlet frit blockage, high fuel temperatures, frit failure, enhanced fission product release, reduced operating time, loss of particles	Small-scale nuclear tests as function of temperature, PIE
2. Hydrogen embrittlement of hot frit or outlet channel material	Frit failure, loss of particles, reduced, operating time	Small-scale nuclear tests as a function of temperature, PIE
3. Plugging of holes in frit from fission products or fuel particles	Overheating, failed frit, loss of particles, reduced operating power or time	Small-scale nuclear facility, PIE
4. UC_2 particle diffusion through carbon layers (so-called amoeba migration problem)	Enhanced fission product release, frit blockage, overheating, reduced operating power or time	Small-scale nuclear facility, PIE
5. High temperature, high temperature gradient from overpower conditions or local power peaking	Melting of fuel particles or frit material, failure of coating material, frit blockage, loss of material, enhanced fission product release, reduced operating power or time	Small-scale nuclear facility, PIE
6a. Channeling of flow through particle bed	Temperature gradients, overheating, coating failure, frit blockage, frit failure, loss of particles, reduced operating power or time	Small-scale nuclear facility, PIE

TABLE 33. (continued)

Failure Mode Initiating Event		Possible Anticipated Consequences	Optimum Facility For Investigation
6b.	Redistribution of particle bed caused by thermal cycling resulting in non-uniform flow distribution	Temperature gradients, overheating, coating failure, frit distortion, frit blockage, frit failure, loss of particles, reduced operating power or time	Small-scale nuclear facility, PIE
6c.	Non-uniform inlet flow to cold frit	Temperature gradients, overheating, coating failure, frit blockage, frit failure, loss of particles, reduced operating power or time	Small-scale nuclear facility, PIE
7.	Material fatigue and stress-strain problems from thermal cycling	Frit failure, loss of particles, reduced operating power or time	Small-scale nuclear facility, PIE
8a.	Pressure drop increase causing choked flow	Flow restriction, overheating, coating failure, frit blockage, frit failure, loss of particles, reduced operating power or time, enhanced fission product release	Small-scale nuclear facility, PIE
8b.	Pressure and flow instability problem due to fluctuating supersonic hydrogen velocities at outlet frit openings	Flow restriction, overheating, coating failure, frit blockage, frit failure, loss of particles, reduced operating power or time, enhanced fission product release	Small-scale nuclear facility, PIE
9.	Power transient caused by liquid hydrogen, control system malfunction, or improper startup procedure	Overheating, coating failure, frit blockage, frit failure, loss of particles, reduced operating power or time, enhanced fission product release	Special transient nuclear facility, PIE
10a.	Flow reduction or no flow at full power	Overheating, coating failure, frit blockage, frit failure, loss of particles, reduced operating power or time, enhanced fission product release	Small-scale nuclear facility, PIE
10b.	Failure to remove stored and decay heat after operation	Overheating, coating failure, frit blockage, frit failure, loss or particles, reduced operating power or time, enhanced fission product release	Small-scale nuclear facility, PIE

TABLE 33. (continued)

Failure Mode Initiating Event	Possible Anticipated Consequences	Optimum Facility For Investigation
11. Problem with integrity of end piece material	Loss of particles, reduced operating time	Small-scale nuclear facility
12. Radiation or thermal stress problems with integrity of moderator-reflector material	Loss of reactivity, reduced operating time	ATR, electrical heating, full-scale engine test
13. Thermoelectric corrosion between frit materials or structural material	Loss of integrity of frit or structural members, reduced operating time	Non-nuclear electric heating
14. Thermal expansion from 4K to $\sim$3000K	Structural failure, flow distribution problems, overheating, loss of material, reduced operating time	Non-nuclear tests, integral engine test
15. Not able to shut down or startup reactor because of control element or control system malfunction	Loss of maneuverability, fuel failure, deplete hydrogen supply	Non-nuclear testing
16. Sintering of fuel particles, possibly enhanced by irradiation induced diffusion	High fuel temperatures, fuel melting, coating failure, reduced operating time	Small scale nuclear tests as function of temperature and burnup, PIE
17. Loss of hydrogen from hydride moderator by means of defect in surrounding can	Loss of reactivity, reduced operating time	Quality assurance measurements, ATR, full-scale engine test

or just crack and remain around the fuel particle, then the consequences of coating failure would be considerably reduced.

3.3 Fuel Qualification Testing Plan

The proposed initial operating goal for a nuclear rocket engine has been to provide approximately 10,000-30,000 lb of thrust at high temperature (2700 to 3000 K), to cycle between low and high temperatures approximately 160 times, with operating times during each cycle of from 2 to 8 min, for a total expected operating lifetime of approximately 10 hr.

No nuclear fuel has ever been subjected to such severe operating conditions in a hydrogen environment. Both the NERVA and PBR fuel designs have been proposed as being potentially capable of performing as required under these severe operating conditions. The purpose of this section is to describe a nuclear fuel testing plan that will provide the opportunity for each fuel design to demonstrate its operational capabilities.

It is assumed that as a part of the fuel manufacturer's qualification testing program, both fuel designs will already have been tested in a hydrogen environment at high temperatures with electrical heating techniques. Since electrical heating cannot simulate fission heat transfer characteristics, temperature distribution, and radiation effects, nuclear facilities capable of testing individual and/or small clusters of fuel elements of each design, or small nuclear reactors made up entirely of each fuel element design (such as the NF-1), will be required.

For the initial development testing to evaluate the effects of high temperature, radiation, and the hydrogen environment, it would be highly desirable from a cost and schedule standpoint to do the testing on subassemblies in a test loop-driver core facility. A test-loop driver core facility has the advantages of producing higher thermal flux and fission power in the test fuel than in the core fuel and the capability of testing one or more fuel elements to failure without damaging the core fuel elements. In order to provide the high power densities and fuel temperatures necessary to evaluate the fuel in the loop, the driver core

fuel may also have to be of the test fuel design, or the entire core may have to be a "test core" without a test loop, such as was the case with the Nuclear Furnace used at the end of the NERVA program. This is discussed further in Section 3.4.

The primary objectives of the proposed test program will be to determine, for each fuel design, the maximum operating temperature conditions that the fuel can successfully operate at for at least 500 cycles (between approximately room temperature and the maximum temperature), and to determine the fuel failure temperature. In addition, limited separate effects tests are proposed to evaluate the effects on fuel behavior of fuel heatup rate (power ramp rate), rapid shutdown (such as Scram from high power, high temperature), flow reduction, and complete loss of coolant (H_2) flow.

3.4 Driver Core Facility Test Program

The proposed test program for subassembly testing both types of fuel concepts in a test loop with a "driver core" is described in this section and summarized in Table 34.

3.4.1 System Checkout Tests. The reactor system and loop need to be operated and checked out over the range of expected operating conditions. Data acquisition and recording capabilities will be evaluated and any identified problems in the systems will be corrected. Power calibration and the ratio between test fuel and core power relationship will be determined and evaluated. In general, the test loop and driver core will be qualified for fuel testing.

3.4.2 Testing to 2600 K

- With the test fuel element(s) in the loop, bring the driver core to critical, establish desired coolant conditions in the loop.

TABLE 34. DRIVER CORE TEST PROGRAM

Test ID (Section No.)	Temperature Ramp Rate (K/s)	Temperature Range (K)	Time at Maximum Temperature (min)	Number of Temperature Cycles	Reactor Shutdown Mode
4.1.2 (new fuel)	40-50	400 to 2000	10	--	--
		2000 to 2100	10	--	--
		2100 to 2200	10	--	--
		2200 to 2300	10	--	--
		2300 to 2400	10	--	--
		2400 to 2500	10	--	--
		2500 to 2600	10	--	Scram
4.1.3 (new fuel)	40-50	400 to 2000	10	--	--
		2000 to 2100	10	--	--
		2100 to 2200	10	--	--
		2200 to 2300	10	--	--
		2300 to 2400	10	--	--
		2400 to 2500	10	--	--
		2500 to 2600	10	--	--
		2600 to 2700	10	--	--
		2700 to 2800	10	--	--
		2800 to 2900	10	--	Scram
		2900 to 3000			
4.1.4 (new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to 400 to 2600	2-8	160	Manual rundown
(new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to 2700	2-8	--	--
		2700 to 400 to 2700	2-8	160	--
(new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to 2800	2-8	--	--
		2800 to 400 to 2800	2-8	160	Manual rundown
(new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to 2900	2-8	--	--
		2900 to 400 to 2900	2-8	160	Manual rundown
4.1.5 (new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to 3000	2-8	--	--
		3000 to 400 to 3000	2-8	500	Manual rundown

TABLE 34. (continued)

Test ID (Section No.)	Temperature Ramp Rate (K/s)	Temperature Range (K)	Time at Maximum Temperature (min)	Number of Temperature Cycles	Reactor Shutdown Mode
4.1.6 (new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to 3100	2-8	--	--
		3100 to 400 to 3100	2-8	160	Scram
	70-80	400 to 2600	As Req'd	--	--
		2600 to 3200	2-8	--	--
		3200 to 400 to 3200	2-8	160	Scram
		(Continue procedure at 100 K increments until failure occurs)			
4.1.7 (new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to (Failure-100)	2-8	--	--
		(Failure-100) to 400 to (Failure-100)	2-8	500	Manual rundown
Repeat procedure 3 more times, each time with new fuel if failure does not occur. If fuel failure does occur proceed as follows:					
(new fuel)	70-80	400 to 2600	As Req'd	--	--
		2600 to (Failure-200)	2-8	--	--
		(Failure-200) to 400 to (Failure-200)	2-8	500	Manual rundown
Repeat procedure 3 more times, each time with new fuel if failure does not occur. If fuel failure does occur, decrease temperature by 100 K until 500 cycles can be performed without failure. Temperature limit will be defined as "Maximum Safe Operating Temperature" or (MSOT).					
4.1.8 (new fuel)	70-80	400 to MSOT to 400	8	50	Scram
	80-90	400 to MSOT to 400	8	50	Scram
	90-100	400 to MSOT to 400	8	50	Scram
	100-110 (etc)	400 to MSOT to 400	8	50	Scram
Continue this procedure at increments of 10 K/s until fuel failure occurs or core limited power ramp rate is achieved					
4.1.9	70-80	400 to MSOT	As Req'd	--	Scram
Loss of hydrogen flow test; hydrogen flow will be resumed after specified time interval. Test would be repeated for longer times if fuel failure does not occur.					

- Increase driver core power at a ramp rate sufficient to produce an approximately 40-50 K/s temperature ramp rate in the test fuel. Terminate temperature increase at a fuel average temperature of ∿2000 K.

- Maintain test fuel temperature at ∿2000 K for 10 min.

- Increase test fuel temperature at same ramp rate to ∿2100 K, then hold test fuel at ∿2100 K for 10 min.

- Repeat previous step and increase test fuel in ∿100 K increments holding at each temperature increase for 10 min before proceeding to the next highest temperature, up to a maximum of ∿2600 K.

- Following operation at test fuel temperature of ∿2600 K for 10 min, shut down the driver core by means of a reactor Scram.

- Remove the test fuel and transport to the hot cell for examination.

3.4.3 Testing to Temperature 3000 K

- Replace the removed test fuel with fresh test fuel of the same design.

- Repeat 3.4.2 but increase the maximum temperature to ∿3000 K (with intermediate holds at 2700, 2800, and 2900 K) instead of ∿2600 K. However, if potential or probable fuel failure of any type is indicated prior to attaining an average test fuel temperature of ∿3000 K, terminate the temperature increase, shut down the driver core and loop, and remove the test fuel for detailed examination.

- If the maximum temperature of 3000 K was attained with no indication of fuel failure, maintain temperature at 3000 K for

10 min and then shut down the driver core in as rapid a manner as possible to evaluate the effect of rapid shutdown on fuel behavior.

- Remove test fuel for detailed examination.

3.4.4 Repetitive Testing to 3000 K

- Replace the removed test fuel with fresh test fuel of the same design.

- Bring the driver core to critical, establish desired inlet conditions in the loop.

- Increase driver core power at a ramp rate sufficient to produce a rapid rate of temperature increase in the test fuel (ramp rate of 70 to 80 K/s).

- Increase the test fuel average temperature ~2600 K, and maintain a steady power level to build up a fission product inventory (time at power will depend on fission product detection instrumentation sensitivity).

- Begin cycling the test fuel temperature between high (~2600 K) and low (~400 K) temperature, 160 times. When at the high temperature during each cycle, maintain the temperature for periods varying (randomly) between 2 and 8 min for each cycle. Shut down the reactor by means of a manual control element run down for all cycle tests, unless a Scram is required.

- After 160 cycles have been completed (assuming no indicated fuel failure, or other problems, during this time) shut down the driver core and remove the test fuel for detailed examination.

- Replace the test fuel with fresh test fuel of the same design.

- Repeat the above procedure at 100 K increments up to 3000 K.

3.4.5 Repetitive Testing at 3000 K (500 cycles)

- Assuming no fuel failures occurred during 3.4.4 above, replace the test fuel with fresh test fuel of the same design.

- Repeat the procedure described for 3.4.4, except increase the maximum temperature to 3000 K from 2600 K and increase the number of cycles from 160 to 500.

- If no failure or other severe problems have occurred during the 500 cycles, shut down the driver core and remove the test fuel for detailed examination.

3.4.6 Fuel Failure Tests

- Replace the test fuel with fresh test fuel of the same design.

- Increase driver core power at a ramp rate sufficient to produce a rapid rate of temperature rise in the test fuel (ramp rate 70 to 80 K/s), and maintain temperature ramp until the fuel average temperature reaches ∿2600 K.

- Maintain test fuel temperature approximately steady at 2600 K and maintain the reactor power level as required to build up a fission product inventory.

- Increase the fuel temperature from 2600 to 3100 K at 70-80 K/s.

- Begin cycling the test fuel temperature between high (∿3100 K) and low (∿400 K) temperature, 160 times. When at the high temperature during each cycle, maintain the temperature for periods varying (randomly) between 2 and 8 min for each cycle.

Shut down the reactor by means of a Scram at the end of each high temperature operation.

- After 150 cycles have been completed, assuming no indicated fuel failure, shut down the driver core by means of a Scram. Remove test fuel for detailed examination. Replace test fuel with fresh fuel of same design and repeat at a fuel temperature 3200 K. Continue this procedure (at 100 K increments) until failure occurs.

3.4.7 Repeatability Tests

- If 3.4.6 is successfully completed, the threshold temperature for fuel failure will have been determined to be $\geq$3100 K. Once the threshold temperature has been determined, replace the test fuel with fresh test fuel of the same design, increase the temperature of the new test fuel to 2600 K, and then increase temperature to the (threshold for failure-100 K), and cycle, for 500 cycles.

- If the test fuel indicates no failure after 500 cycles, shut down and remove test fuel for examination. Replace the test fuel with fresh fuel of the same design and repeat the 500 cycles at the same test fuel temperature conditions. Repeat this last step three times to demonstrate that the indicated (threshold for failure -100 K) is indeed a safe temperature at which the fuel can be operated without expected failures.

- If the test fuel failed during the first 500 cycles at (threshold of failure -100 K) temperature, replace the test fuel with fresh fuel of the same design, increase the test fuel temperature to (threshold of failure -200 K) and cycle for 500 cycles, as previously described with three repetitions with new fuel for each attempt. Continue this procedure until a fuel temperature is determined at which tne fuel design can successfully be cycled 500 times without expected failure (defined as maximum safe operating temperature).

In addition to the tests described in 3.4.1 through 3.4.4 to evaluate the power and temperature conditions for fuel failure and for evaluating repeatability, some separate effects tests have been identified that can be performed in the test loop-driver core facility.

3.4.8 Fuel Temperature Ramp Rate Tests

Although 3.4.2 through 3.4.7 above describe tests involving test fuel temperature ramp rates of 40-50 K/s and 70-80 K/s, it is not expected that fuel failure should occur at these ramp rates. A series of ramp rate tests should be performed during which the test fuel temperature is ramped from $\sim$400 K to the previously identified "maximum safe operating temperature" and held at that temperature for approximately 8 min, and rapidly returned to room temperature by means of a Scram (with 50 cycles at each ramp rate). The test will start with fresh fuel elements. For this test series, the initial ramp rate should be 70-80 K/s, with the ramp rate increasing by 10 K/s for each succeeding 50 cycles (e.g., the succeeding steps would be 70-80 K/s, 80-90 K/s, 90-100 K/s) until the limit of the driver core is reached or indicated fuel failure occurs.

3.4.9 Loss-of-Flow Testing

Loss-of-flow testing will conclude the test program for each fuel concept. Fresh fuel elements will be operated at the maximum safe operating temperature defined in 3.4.7 for a total time calculated to produce a fission product inventory equivalent to the maximum decay heat source expected from nuclear rocket fuel operation. After the specified fission product inventory has been generated, the hydrogen coolant flow will be shut off completely and the reactor scrammed automatically. The temperature of the test fuel elements and restraining hardware will be monitored for a specified time interval. At the end of this time interval or if abnormally high temperatures are reached, the hydrogen coolant flow will be reinstated. Depending on the results of the test, the test would be repeated with new fuel elements, but extending the time duration prior to turning on the hydrogen flow.

3.5 Small Reactor Test Program

Detailed reactor physics and thermal calculations will be necessary to determine if it is feasible to construct a test loop and driver core facility that will produce sufficiently high power to test both fuel concepts. If a test loop and driver core facility is not feasible or not built for other reasons, then testing in a small test reactor facility with a core composed of all NERVA-type or all particle bed reactor-type fuel elements should be performed. This was the approach used by Los Alamos National Laboratory in 1972 to test nuclear rocket fuel elements. The test facility, called Nuclear Furnace-1, consisted of 49 NERVA-type fuel elements which produced a peak power of 44 MW. The reactor was mounted on a movable railcar so that the reactor could be moved to a hot cell for posttest examination. The Nuclear Furnace-1 facility consisted of two components: a permanent portion including the reactor reflector and control elements; and a replaceable portion consisting of the moderator and fuel elements and associated components.

Two different reactor cores, one using 1.32-m-long NERVA-derivative fuel elements and the other using 0.73-m-long PBR-type fuel elements, will be necessary. It would be extremely advantageous if one or more of the core fuel elements could be readily removed for examination and replaced with fresh fuel after each phase of the test program where temperatures exceed $\sim$2000 K. Since it may not be possible to include this quick removal and replacement design feature, this test plan will assume that the entire core must be shipped to a hot cell for examination and reassembly.

A test plan for qualifying each fuel concept in a nuclear furnace-type facility is given in this section and summarized in Table 35. The test plans for the two fuel concepts are assumed to be the same.

3.5.1 System Checkout Tests

The operating characteristics of the test reactor will be determined after verifying that the following reactor systems are functioning properly:

TABLE 35. SMALL REACTOR TEST PROGRAM

Test ID (Section No.)	Temperature Ramp Rate (K/s)	Temperature Range (K)	Time at Maximum Temperature (min)	Number of Temperature Cycles	Reactor Shutdown Mode
4.2.3	40-50	400 to 1000	As Req'd	--	--
	40-50	1000 to 2000	10	--	--
		2000 to 2100	10	--	--
		2100 to 2200	10	--	--
		2200 to 2300	10	--	--
		2300 to 2400	10	--	--
		2400 to 2500	10	--	--
		2500 to 2600	10	--	--
		2600 to 2700	10	--	--
		2700 to 2800	10	--	--
		2800 to 2900	10	--	--
		2900 to 3000	10	--	--
4.2.4	70-80	400 to 2500 to 400	2-8	50	Manual rundown
	70-80	400 to 2600 to 400	2-8	50	Manual rundown
	70-80	400 to 2700 to 400	2-8	50	Manual rundown
	70-80	400 to 2800 to 400	2-8	50	Manual rundown
	70-80	400 to 2900 to 400	2-8	50	Manual rundown
	70-80	400 to 3000 to 400	2-8	50	Manual rundown
If no fuel failure occurs during 4.2.3 or 4.2.4, proceed as follows:					
4.2.5	70-80	400 to 3000 to 400	2-8	500	Manual rundown
If fuel failure occurred during 4.2.3-4.2.4, proceed as follows:					
4.2.5	70-80	400 to (Failure-100) to 400	2-8	500	Manual rundown
4.2.6	70-80	MSOT	As Req'd	--	Scram

Loss of hydrogen flow test; hydrogen flow will be resumed after specified time interval. Test would be repeated for longer times prior to flow resumption if fuel failure does not occur.

- o Control system
- o Moderator cooling system
- o Hydrogen supply system
- o Hydrogen cooldown and cleanup and disposal system
- o Reactor instrumentation
- o Data acquisition, processing, and display system
- o Emergency shutdown and cooling systems.

3.5.2 Nuclear Operation Checkout

After demonstrating that the reactor systems are functioning properly, the reactor will be taken critical and low power physics measurements performed in the following areas:

- o Neutron instrumentation calibration
- o Neutron flux measurements
- o Reactor shutdown reactivity
- o Reactor excess reactivity
- o Control element calibration
- o Temperature coefficient of reactivity for moderator and fuel
- o Power stability measurements.

Results of all these measurements will be compared with reactor physics calculations to bench-mark the model for high-temperature operation.

3.5.3 Testing to 3000 K

After the low power physics measurements have been completed, testing to 3000 K will be initiated.

- Increase reactor power gradually to obtain exit gas temperature of 1000 K.

- Verify that all systems and instruments are functioning properly. Shut down the reactor if fuel failure or any other operating anomalies are observed during this or any of the following steps.

- Increase reactor power to obtain exit gas temperature ramp increase rate of 40-50 K/s up to 2000 K.

- Maintain exit gas temperature of 2000 K for 10 min.

- Increase exit gas temperature by 100 K to 2100 K over a 10 s time span and then maintain 2100 K for 10 min.

- Repeat this procedure at 100 K increments until reaching 3000 K. If measurements indicate probable fuel failure at any time, the reactor will be shut down immediately.

- If fuel failure occurs, the reactor core fuel elements will be examined and replaced as necessary.

- After the reactor is reconfigured, the reactor exit gas temperature will be limited to 100 K below the previous failure temperature.

3.5.4 Repetitive Testing up to 3000 K

The anticipated duty cycle of nuclear rocket fuel consists of numerous short duration operations of 2 to 8 min at maximum permissible

temperature. The cycle testing will define this maximum temperature limit. If measurements indicate probable fuel failure has occurred, the reactor will be shut down immediately.

- o Establish reactor critical conditions at low power.

- o Increase the gas exit temperature at a rate of 70 to 80 K/s up to 2500 K.

- o Begin cycling the test fuel temperature between 2500 K and ∿400 K 50 times with varying random hold times from 2 to 8 min.

- o Shut down the reactor at the end of each hold period by means of a manual control element insertion, unless a reactor scram is required.

Repeat this procedure at 100 K increments up to 3000 K, with 50 cycles at each temperature.

3.5.5 Repetitive Testing at 3000 K (500 cycles)

Assuming no fuel failures have occurred during the previous testing, the fuel will be tested at 3000 K for 500 cycles. If fuel failures have occurred, the reactor fuel elements will be examined and replaced as necessary. After the reactor core is reconfigured, the reactor exit gas temperature will be limited to 100 K below the previous failure temperature.

- o Establish reactor critical conditions at low power.

- o Increase the gas exit temperature at a rate of 70 to 80 K/s up to 3000 K.

- o Begin cycling the test fuel temperature between 3000 K and ∿400 K for 500 times with randomly varying hold times of 2 to 8 min.

- Shut down the reactor at the end of each hold period by means of a manual control element insertion, unless a manual scram is required.

- If fuel failure occurs, the reactor will be shut down immediately. The fuel elements will be examined and replaced as necessary.

3.5.6 Loss-of-Flow Testing

Loss-of-flow testing will conclude the test program for each fuel concept. The reactor will be operated at the maximum safe operating fuel temperature defined in 3.5.5 for a total time that is calculated to produce a fission product inventory equivalent to the maximum decay heat source expected from actual nuclear rocket fuel operation. After the specified fission product inventory has been generated, the hydrogen coolant flow will be shut off completely and the reactor scrammed automatically. The reactor will be permitted to heat up for a specified time interval and then the hydrogen coolant flow will be reinstated. The heatup time duration will depend on detailed analysis and electrical heating test results. The test would be repeated for longer time durations prior to flow resumption if fuel failure does not occur. It is expected that optical pyrometer measurements will be required to monitor the temperature of the reactor.

3.5.7 Summary of Small Reactor Test Program

Detailed examination of the reactor core components would be completed after each core removal and following completion of the program. Failure testing to temperatures above 3000 K was not included in the test program because of the cost and time delay in removing, examining and reconfiguring, and decontaminating the facility after fuel failure. The severity of the whole-core small reactor test program has to be moderated with respect to the test loop-driver core test program in order to minimize the frequency and extent of fuel replacement and facility decontamination.

3.6 Facility Specification and Requirements

The objective of this section is to describe the functions, the design requirements, and the specifications for a facility to qualify nuclear rocket fuel.

3.6.1 Facility Functions

The facility will provide the following functions for testing and qualifying fuel elements for space reactor use:

- High neutron flux for testing one or more fuel elements at required boundary conditions
- Electrical power for a variety of equipment
- Containment building for reactor and test loop
- Heat removal system for reactor fuel
- Hydrogen storage and supply at required conditions
- Inert gas storage and supply for inerting the test loop
- Inlet and outlet piping for various gases and liquids of loop and plant
- System for acquiring, displaying, recording, and processing data from test fuel, loop, and plant instruments
- Control system for operating loop and plant remotely
- Crane for lifting test assembly, casks, and other equipment
- Shielding for protecting personnel from radioactivity

- Emergency equipment for personnel and plant protection

- Cooldown, cleanup, and disposal (or recycling) of effluent gases from test fuel elements

- Posttest storage of fuel elements

- Security protection of plant and highly-enriched uranium

- Office space for operating personnel.

3.6.2 Design Requirements

The general design requirements for a facility to test both NERVA derivative and particle bed reactor fuel elements are listed in this section. This is followed by a discussion of possible concepts for such a facility and a preliminary evaluation of existing facilities.

The basic design requirements for the nuclear rocket test facility are listed below.

1. Capability to test during steady-state conditions one or more NERVA or particle bed fuel elements up to temperatures of at least 3000 K with typical hydrogen and power density conditions.

2. An effluent cleanup system will cool the effluent gas and remove sufficient radioactive contaminants to permit either discharging the hydrogen into the atmosphere via flaring or recirculating the hydrogen via a closed loop design.

3. The hydrogen supply system will provide hydrogen gas flow at pressure up to 14 MPa and flow rate up to 50 g/s per NERVA element for at least 4-hr continuous operation. Hydrogen inlet temperature will be at $\sim$350 K. Hydrogen gas will also be provided for decay heat removal. The hydrogen flow rate per

particle bed fuel element is not known exactly but is expected to be ~300 g/s.

4. Hydrogen supply system will be operable both directly from the reactor building and remotely from the control system. Manual and computer modes of hydrogen control will be provided from the control room.

5. Reactor and hydrogen supply system will be capable of operating at 100% power continuously for at least 4 hr.

6. Active fuel length of reactor will be ~1.32 m. The active length of the PBR fuel is expected to be ~0.50 m, with an additional ~0.12 m of material at each end of the fuel.

7. The reactor will have a negative temperature coefficient of reactivity.

8. The reactor will have adequate shutdown reactivity with test fuel elements installed.

9. The coolant effluent from reactor and test fuel elements will be monitored for presence and magnitude of fission products to detect fuel failure. Reactor will be automatically shut down at preset level of activity.

10. The reactor control room will either be located a safe distance (based on calculated dose rate) from the reactor or designed to protect operating personnel while performing tests with maximum expected failure consequences.

11. System(s) for removing decay heat from reactor fuel and test fuel elements will be provided.

12. A containment building will house the reactor and will be designed to prevent major releases of radioactivity into the atmosphere in the event of an accident.

13. The reactor will be automatically shut down in the event of abnormal hydrogen flow to test fuel elements or to reactor fuel elements.

14. The reactor control system will provide for both manual and computer-controlled operation.

15. The reactor control system will provide for both fast shutdown (within 2 s) and rapid rundown of reactor power (within 20 s) for preset alarm levels of selected reactor and test fuel element instrumentation.

16. Provision for measurement and recording data of fuel elements, temperature, pressure, coolant flow rate, differential pressure, inlet and outlet coolant temperature, and neutron flux.

17. An emergency system for cooling the reactor and test fuel element in case the normal cooling system(s) malfunction.

18. Adequate shielding of radioactive material to protect personnel.

19. Systems for inerting the fuel elements and reactor region with either helium or nitrogen gas after nuclear operation.

3.6.3 Test Reactor Concepts

Detailed feasibility studies will have to be completed to determine if a Space Test Reactor (STR) should be a test loop-driver core-type or a nuclear furnace-type design. The test loop-driver core concept would probably consist of a water-moderated, helium-cooled core of NERVA-type or PBR-type fuel elements, a central in-pile tube in which a single or a small cluster of test fuel elements is located, and a loop coolant system for

hydrogen gas cleanup and recirculation. The driver core fuel elements would be separately cooled by helium to minimize corrosion effects and hydrogen explosion problems. The helium flow rate would be large to minimize core fuel temperatures while still operating at extremely high power densities.

This concept has the following advantages: able to test single fuel elements; adequate power density capability; minimum amount of hydrogen for cleanup and recirculation used; minimum amount of radioactive material to be cleaned up after fuel failure.

The disadvantages appear to be the following: separate helium coolant system required; fuel development required; repetitive testing of different single fuel elements required to establish confidence in data; and operating lifetime of driver core not known.

The other reactor concept would be patterned after the Nuclear Furnace-1 reactor that consisted of 49 NERVA fuel elements that were water-moderated and cooled with hydrogen gas. The reactor operated at $\sim$2450 K for a total of $\sim$109 min during six runs. The reactor was then disassembled in a hot cell for fuel and component examination. An effluent system cooled and removed much of the radioactive material in the hydrogen gas before disposal to the atmosphere by flaring. Since both NERVA and particle bed fuel assemblies are planned to be tested, two separate reactor core designs would be required.

This concept has the following advantages: large number of fuel elements are tested at same time, for improved statistics; and the power capability is adequate.

This concept has the following disadvantages: large amount of hydrogen gas required because recirculation would probably not be practical; fuel development required; expensive and time consuming to replace entire core after each test phase; cleanup of radioactive material could be severe problem after extensive fuel failure; and separate reactor

core designs of NERVA-type and particle bed type fuel elements would be expensive.

3.6.4 Test Loop Design Requirements

If a test loop-driver core concept is selected, the following design requirements should be addressed:

1. The inpile tube will be capable of withstanding maximum pressure of at least 20 MPa.

2. The inpile tube will be of sufficient size to accommodate a cluster of seven NERVA-type fuel elements ($\sim$6 cm overall diameter) or a single particle bed fuel assembly ($\sim$6.5 cm diameter).

3. The fuel lengths of the two fuel elements will be addressed in the design, 1.32 m for NERVA and $\sim$0.73 m for a PBR fuel assembly which includes the length of insulating material adjoining the active fuel length.

4. The inpile tube design will provide for installing at least 80 pairs of instrumentation leads to measure fuel element temperatures, hydrogen temperatures and pressures, differential pressures and temperatures, neutron flux, and other miscellaneous measurements.

5. The loop system will provide hydrogen gas flow at controllable pressures and flow rates. Pressures will be variable up to 14 MPa. Flow rate will be variable up to 400 g/s.

6. Hydrogen coolant system will have capability for providing up to 400 g/s flow at pressures up to 14 MPa for time periods up to 4 hr.

7. The hydrogen cooling flow per fuel element will be typical (approximately 50 g/s for each NERVA-type fuel element and ∿300 g/s for a PBR fuel assembly).

8. Hydrogen gas inlet temperature will be approximately 350 K and the outlet temperature will be up to 3200 K.

9. Hydrogen cooling system for test fuel elements will be separate from driver core cooling system.

10. Loop hydrogen flow system will be operable both directly from the reactor building and remotely from the control room. Manual and computer modes of hydrogen control will be provided from the control building.

11. A water injection system, similar to that employed in the Nuclear Furnace-1 Test, will be provided to initially cool the effluent hydrogen flow to an acceptable temperature for the outlet piping.

12. An effluent cleanup system will be designed to remove sufficient radioactive contaminants to permit either discharging the hydrogen into the atmosphere via flaring or, preferably, recycling the hydrogen via a closed loop design.

13. An emergency cooling system will be provided to supply short term cooling for the test fuel elements in case the hydrogen cooling system malfunctions.

14. Reduced gas flow rate will be provided for decay heat removal.

15. Helium and nitrogen gases will be provided for inerting the test loop and fuel after nuclear operation.

16. Coolant effluent from the test fuel elements will be monitored for the presence and magnitude of fission product activity. The reactor will automatically shut down at a preselected level.

17. Adequate shielding to attenuate residual radioactivity to permissible levels will be included in the design.

3.6.5 Evaluation of Facilities for Space Reactor Fuel Testing

The Power Burst Facility (PBF), the Engineering Test Reactor (ETR), the Transient Reactor Test Facility (TREAT), the High Flux Isotope Reactor (HFIR), and the Advanced Test Reactor (ATR) were evaluated for their potential in testing nuclear rocket fuels. The PBF is a driver core-type design reactor that was used for many light water reactor fuel safety tests until February 1985. It operated at a maximum steady-state power of 26 MW and a maximum transient power of $\sim$60,000 MW. The active core length is 0.91 m, considerably shorter then the 1.32 m length required for testing full-length NERVA-type fuel elements. Based on experience with previous PBF fuel testing results, a single NERVA-type fuel element would operate at only about 125 kW at a PBF power of 26 MW. A steady-state fuel element power of $\sim$1.5 MW is required for reaching high temperatures at typical hydrogen flow rates. During a short-period transient, the PBF could drive a single NERVA fuel element up to a peak power of $\sim$300 MW for a very short time. The energy thus produced in a NERVA fuel element in a PBF power transient would probably be less than that required for failure of the NERVA fuel.

Before the ETR was shutdown, it operated at a maximum power level of 175 MW for material irradiation effects studies. The reactor active fuel length was 0.91 m, about 0.41 m shorter than needed to test NERVA-type fuel elements. The estimated maximum power achievable for a single NERVA-type fuel element in ETR is 700 kW. Considerable effort would be required to increase the core active length to 1.32 m, but is believed to be feasible. Much of the peripheral support equipment has been dismantled, so considerable expense would be necessary to rejuvenate the facility.

The ATR is capable of operating up to 250 MW with a power density of 1 MW/L at a maximum thermal neutron flux of 1×10^{15} n/cm$^2\cdot$s. The active fuel length of the ATR is 1.22 m, which is about 0.1 m less than the length of a NERVA-type fuel element. A scoping-type reactor physics

calculation was performed to determine the maximum power achievable for a single NERVA-type fuel element in ATR. The results of this scoping calculation (results are given in Appendix J) indicate that a maximum power of $\sim$1 MW could be obtained for a single NERVA-type fuel element (with a fuel loading of $\sim$160 g ^{235}U) tested in ATR. This power would be sufficient to produce a hydrogen exit temperature of $\sim$3000 K for a hydrogen flow rate of $\sim$25 g/s (neglecting heat losses which would probably be on the order of 10 to 20%). The average hydrogen flow rate per fuel element in the Nuclear Furnace-1 facility was $\sim$22 g/s (the peak flow for the highest power fuel elements was $\sim$40 g/s per fuel element). Thus, it may be feasible to test NERVA-derivative fuel to $\sim$3000 K in ATR, but not for exactly typical flow rates or for full-length heating. A higher power may be achievable if a special, new design driver core made with ATR-type fuel elements were used with an improved thermal-hydraulic design for cooling the core fuel elements.

The maximum power achievable for a single PBR-type fuel assembly in a driver core made of ATR fuel was not calculated, but is estimated (on the basis of fuel content and length) to be $\sim$1.5 MW. A typical power of $\sim$10.5 MW per fuel element is needed to simulate a typical fuel element in a 200 MW particle bed reactor with a hydrogen flow of 5 kg/s and an outlet temperature of 2750 K. Therefore, a driver core made of ATR fuel assemblies would be inadequate for nuclear rocket fuel testing of particle bed fuel elements.

Since the ATR is funded primarily by the U.S. Navy for nuclear fuel studies, it would be difficult to perform short-duration irradiation testing to expected fuel failure. Relatively low-temperature steady-state testing for fuel development purposes could be performed in ATR.

The Transient Reactor Test Facility (TREAT) was also considered for possible nuclear rocket fuel testing. Since the reactor only has a maximum steady-state power capability of 120 kW, steady-state testing is definitely not possible. Transient testing of nuclear rocket fuels for very brief irradiation periods and for less than failure conditions would probably be possible.

The High Flux Isotope Reactor (HFIR) was also considered since it has a high thermal flux capability, but the reactor does not include provision for a test loop and the active fuel length is only 0.813 m.

3.6.6 Non-Nuclear Test Requirements

The non-nuclear tests needed to qualify the fuel, moderator, reflector, and control elements and the control mechanical and electrical systems are briefly described in this section. It is assumed that an extensive non-nuclear electrical heating test program will be performed elsewhere by the fuel vendor. Sufficient electrical heating tests should have been completed to indicate that the fuel will perform to the desired operating goal for rocket fuel. In order to qualify the design of flight-related systems, it will be necessary to validate the system performance of fuel elements, moderator elements, control elements, mechanical devices, and electronic gear in a simulated space environment. The following tests should be performed:

1. One main concern is the possibility of material bonding during long durations of non-operation in vacuum at low temperature.[22] Individual components should be tested prior to, during, and after long periods of storage at expected environmental conditions in space. Testing should be performed for reactor material that has been previously irradiated to expected burnup and fluence magnitudes.

2. Another non-nuclear facility will be needed to perform simulated launch vibration tests of the reactor, engine, and rocket components.

3. Non-nuclear heating and cycling of moderating, reflector, and control elements to simulate expected conditions will be required to qualify these components.

4. Separate qualification testing of the control system mechanical, instrumentation, and electronic components will be required to ensure their operability in space.

5. Non-nuclear testing of particle bed reactor components will be needed to verify that there are no thermoelectric corrosion problems between the frit and end piece materials.

3.7 Discussion and Conclusions

In order to qualify fuel elements for a nuclear rocket engine, extensive testing such as that proposed in this report will be necessary. It is anticipated that the test program will be modified to address particular concerns for each fuel concept after more operating experience is acquired. Because of the high power density of the NERVA-derivative fuel (3.6 kW/cm^3 compared to 0.85 kW/cm^3 for a PWR) and the extremely high power density of the PBR-type fuel (8.2 kW/cm^3), a special, high-flux facility or facilities will be needed to test these fuel elements to 3000 K. Based on a reactor physics scoping calculation made for this report, it appears feasible to test a single NERVA-derivative fuel element up to 3000 K in a loop-driver core made of ATR fuel assemblies. Such a facility, however, could not test PBR-type fuel elements to 3000 K unless the hydrogen flow was reduced to $\sim$15% of normal. Test results performed with such low flow could not be confidently used to qualify fuel elements because of potential erosion, corrosion, and pressure drop problems at higher flow rates.

A problem with non-typical active fuel length would arise if ATR-type fuel assemblies were used to make a driver core because the ATR fuel length is $\sim$10 cm shorter than NERVA-derivative fuel and $\sim$49 cm longer than PBR-type fuel. Testing in a driver core with longer than typical fuel length could produce ambiguous results since the flux and temperature gradients would also be non-typical. Testing in a driver core with shorter than typical fuel length should be avoided, if possible, since the fuel length, the power profile, and the temperature gradients would all be non-typical. Since a new facility appears necessary to test both fuel

concepts, it would be a definite advantage to build a test facility that could operate with the proper fuel length for each fuel concept.

Another important consideration is whether testing a single fuel element is adequate for qualification or whether full-scale reactor core testing will be required. There is no doubt that single-fuel-element testing will not address problems with the structural support members, fuel element interactions, moderator and reflector material, or control system problems. The radiation and temperature-related operating problems, however, should be well-suited for investigating with a single fuel element in a loop. Testing in a loop would require less time and resources and permit more severe tests than could be performed routinely in a whole core-type facility. However, final demonstration testing will have to be performed in a full-scale prototypical engine test.

An instrument evaluation and development program will be necessary to monitor the outlet gas temperature when operating at temperatures approaching 3000 K. If thermocouples were available, it would be worthwhile to also monitor fuel element material temperature.

Assuming that both NERVA and PBR fuel elements will be tested and qualified, the recommended facility is a test loop-driver core made of NERVA-type fuel elements with the capability of switching to a core made of PBR fuel elements if necessary. The test fuel in the loop would be hydrogen-cooled, while the core fuel elements would be helium-cooled. A test loop-driver core facility has the great advantage of being able to test single fuel elements of either type at high temperatures while minimizing the hydrogen cleanup and decontamination problems. Reactor physics and thermal calculations should be performed to determine the feasibility of testing both NERVA and PBR fuel elements in a driver core made of NERVA-fuel.

The disadvantages of this facility are: it may be difficult and expensive to design and construct a dual-core reactor test facility; PBR fuel design (if needed as a driver core) may require considerable

development effort, but the PBR development could proceed in parallel with the construction and testing of the NERVA fuel.

If a test loop-driver core design is not feasible, then the recommended option would be whole-core testing by constructing a nuclear furnace-type facility with interchangeable cores. One core would be made using all NERVA-derivative fuel elements and the other core made using all PBR fuel elements. This option has the advantages of using a proven design for the NERVA type fuel; test results would be for a relatively large number of fuel elements, instead of only one fuel element at a time. The disadvantages are: plan for whole-core testing has to be moderated relative to that for a test loop-driver core program; hydrogen cleanup system may not permit recirculation of coolant; much more expensive to test whole-core; longer time required to accomplish testing; large reactor power required for PBR-fuel design; and PBR fuel development required to construct PBR facility.

Other options that were considered, but not rated as high as these two recommended facilities, were:

1. Separate test loop-driver core facilities for each fuel concept. Two separate facilities would be very useful but too expensive unless on a "crash" program.

2. A test loop-driver core made of 1.32-m-long PBR fuel assemblies would be able to test both fuel types. Such a facility would be very expensive to build and large amount of PBR fuel development would be needed.

3. A full-scale nuclear rocket engine test facility capable of accommodating either fuel concept would eliminate smaller-scale testing and address possible full-scale problems early in the program. This is a very expensive and time consuming way to test two fuel concepts if fuel problems develop at lower than required temperatures. The hydrogen cleanup system may become so contaminated that further testing is delayed or cancelled. A

large reactor restricts testing to expected nonfailure regime and building a PBR reactor using unproven fuel appears especially risky.

4. A facility that is designed to operate with either type of fuel both as a small-scale test facility (50 to 100 MW) for initial testing and also as a full-scale nuclear rocket engine (200 to 400 MW) would eliminate the need for a separate small-scale facility. Such a facility would be very difficult and expensive to build, it would be time consuming to perform all tests in one facility, decontamination of the hydrogen cleanup system may be difficult and time consuming, whole-core test program has to be moderated, and an unproven PBR appears risky.

It should be noted that a test loop-driver core facility designed to test one or more fuel elements for the nuclear rocket program would also be extremely useful for performing burst power testing of candidate thermal reactor fuel elements for the Multimegawatt (MMW) Strategic Defense Initiative (SDI) program. The loop of the nuclear rocket fuel test reactor should be large enough to test any proposed MMW fuel design. The test loop-driver core facility would also be invaluable for fuel development type testing of candidate thermal reactor fuel concepts for either the nuclear rocket or MMW programs. Various design parameters, such as coating material or coating thickness, could be readily tested at any desired temperature.

A great deal of cooperation and coordination between the fuel vendors, the test facility designers, and test program personnel would be needed to design, build, and test either of these reactors and fuels.

4. PRELIMINARY ENGINE SYSTEM ANALYSIS

Rocketdyne performed the preliminary engine study under contract to the Idaho National Engineering Laboratory to provide component performance and cost estimates for a nuclear rocket engine. The engine was tentatively sized at 15K lbm thrust for DTV applications based on preliminary mission analysis work by Martin Marietta.

A preliminary system design was conducted to determine required conditions, estimate engine performance, and provide data for component preliminary designs. Based on this initial engine cycle balance, preliminary turbopump, nozzle, and valve/control system designs were established. Representative component weights, development plans, and schedules were prepared by the cognizant engineering groups within Rocketdyne. In addition, the cost analysis group developed ROM component cost estimates given in representative ranges.

4.1 Engine System

A representative nuclear thermal rocket engine system has been established to provide conceptual designs of the non-nuclear components, component development plans, and ROM costs. As shown in Figure 42, the engine system configuration analyzed was a simple expander cycle. The engine system utilized low pressure liquid hydrogen from cryogenic propellant tanks flowing to a low pressure pump (boost pump) which provides moderate pressure to the high pressure pump (main pump). Both of these pumps are powered by turbines driven by warm hydrogen heated as it passes through the thrust chamber cooling jacket. The thrust chamber (reactor pressure vessel and nozzle) is convectively cooled regeneratively by first cooling the pressure vessel (high heat flux regions) and the subsonic regions of the nozzle followed by the lower heat flux supersonic portion of the nozzle. The heated hydrogen is used to drive the boost and main turbines in parallel. The main propellant valve and a turbine bypass valve are shown in Figure 42.

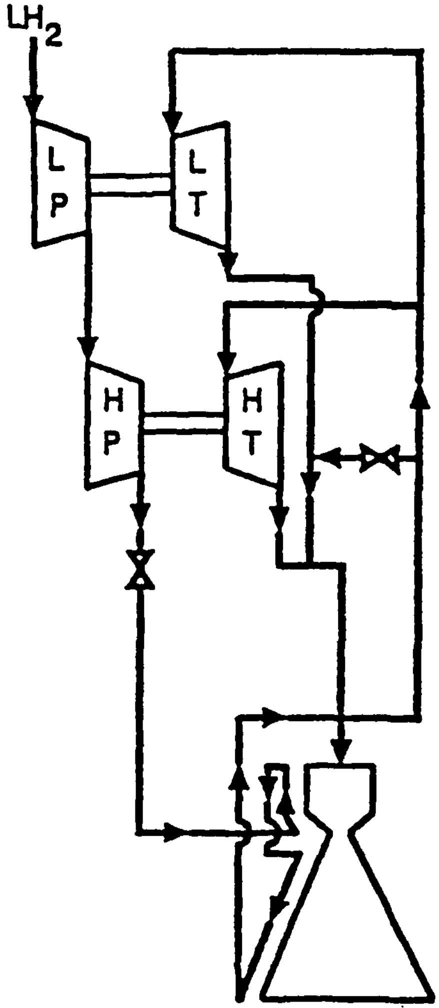


Figure 42. Representative nuclear engine schematic - expander cycle.

4.2 Preliminary Engine Cycle Balance

A preliminary engine cycle balance was performed for the nuclear rocket engine using the Rocketdyne nuclear thermal rocket engine cycle balance computer code. The design requirements for the engine were:

Thrust	:	15,000 lbf
Champer Pressure	:	500 psia
$Q_{Reactor}$	:	330,000 Btu/sec
Nozzle Area Ratio	:	600-to-1
Nozzle % Length	:	80%

The analysis performed balanced the turbine power to drive the pumps, designed the corresponding turbine and pumps, calculated propellant flowrates, pressures, and temperatures within the engine, and determined the engine-delivered specific impulse. These preliminary engine cycle balance results were used to establish the preliminary component designs discussed later.

A maximum hydrogen temperature of 5274°R was achieved at the reactor outlet with a hydrogen flowrate of 16.285 lbm/sec. The assumed 600-to-1 area ratio nozzle engine resulted in a delivered specific impulse of 923 lbf sec/lbm. This delivered specific impulse was determined using the simplified JANNAF methodology and included nozzle geometric boundary layer and reaction kinetic losses.

4.3 Non-Nuclear Components

4.3.1 Turbopumps

Design Description

As shown in Figure 43, the preliminary design for the main turbopump consists of a two-stage centrifugal pump (left side of figure) powered by a single-stage, full admission axial turbine (right side of figure). The preliminary design of the boost turbopump (Figure 44) consists of a single-stage centrifugal pump (left side of figure) driven by a single-stage full admission axial turbine (right side of figure). Both turbines utilize 50% reaction. The turbopump overall dimensions are approximately 7 in. in diameter and 23 in. in length for the main turbopump and 6 in. in diameter and 16 in. in length for the boost turbopump. The boost turbopump incorporates conventional ball bearings; whereas, the main turbopump utilizes a hybrid bearing containing ball bearings and hydrostatic bearings. The ball bearings react radial and axial loads at low speeds and the hydrostatic bearings react radial loads at high speeds. A balance piston on the second stage impeller reacts axial loads at high speeds. To achieve the long life goals of the nuclear rocket engine, further analysis should be performed to trade off a conservative

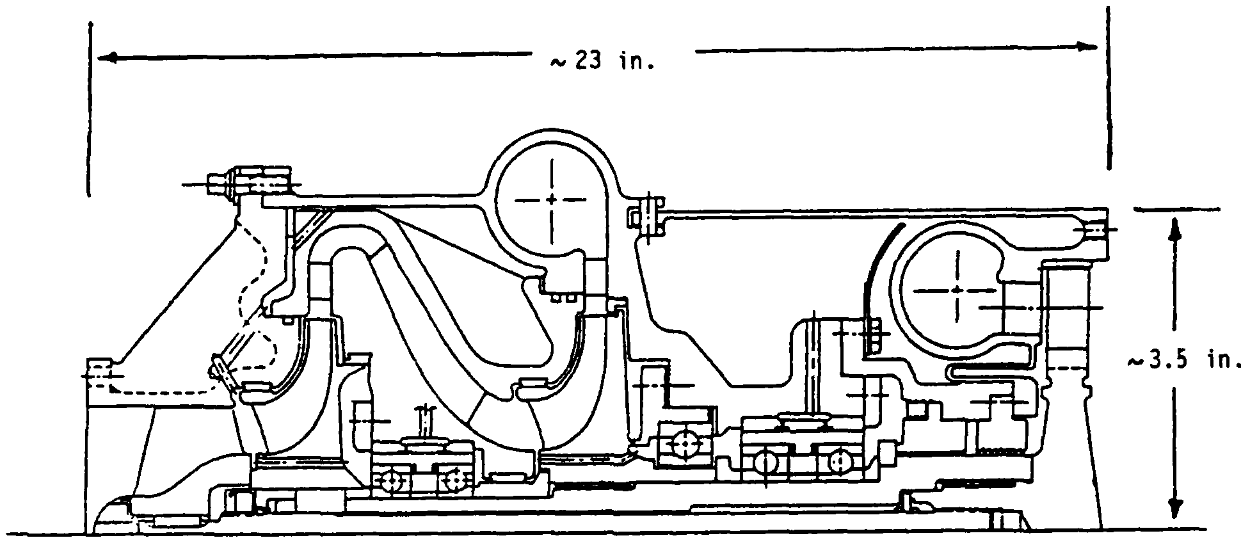


Figure 43. Nuclear engine main turbopump.

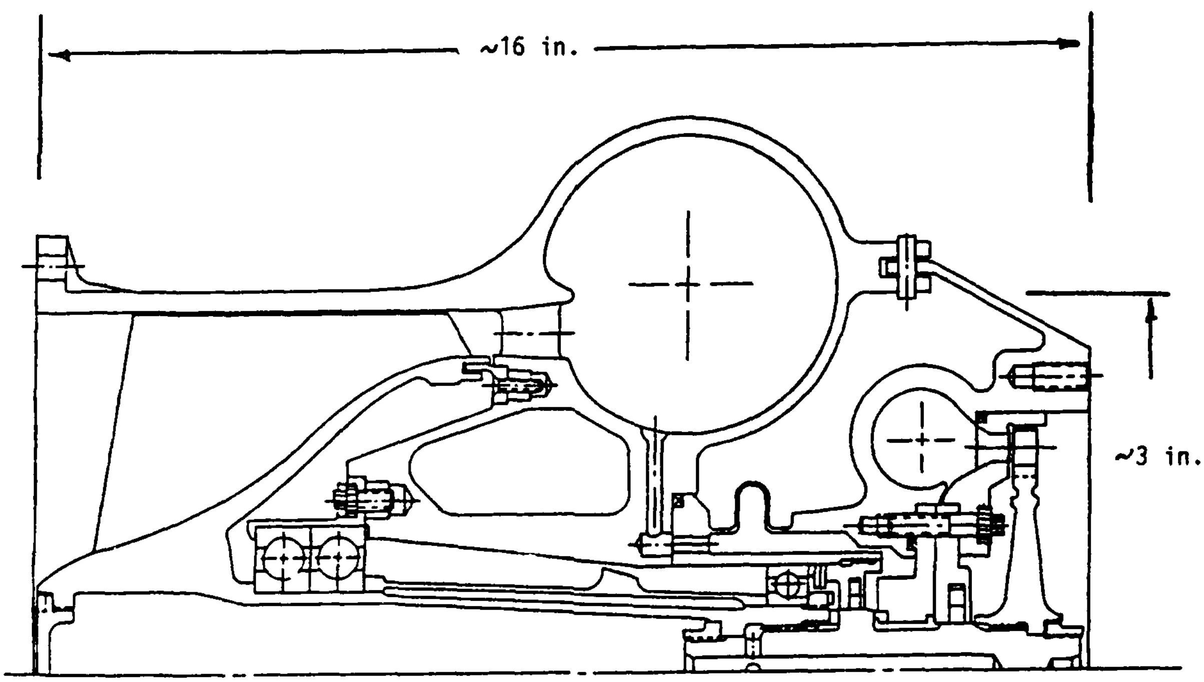


Figure 44. Nuclear engine boost turbopump.

long life design approach with high performance. These studies would include the evaluation of alternative bearing and seal designs. The effect of throttling on axial thrust and the ability to react it within the turbopumps are other issues that needs further evaluation.

The design parameters and operating characteristics for the main and boost pumps and turbines are presented in Tables 36 and 37, respectively. The discharge pressure for the boost pump is 117 psia running at approximately 20,000 rpm. The speed of the main pump was set at 75,000 rpm and provided a pump discharge pressure of 2046 psia.

Development Plan

As shown in Figure 45, a preliminary development plan for the nuclear rocket engine turbomachinery (boost and main turbopumps) result in a 3.5-year program. The turbopump designs through critical design review are completed by the end of the first year. Hardware procurement and fabrication are initiated in the first year and continue through the second year.

The development plan incorporates component testing at Rocketdyne. The turbopump assembly begins at the start of the second year and is completed in the third year as design modifications are required. Test facility preparation is initiated in the second year. To reduce testing and hardware modification costs, pump tests in water and turbine tests using gaseous nitrogen will be conducted to verify component performance and identify potential problems early in the program. As illustrated in Figure 45, the initial turbopump tests with hydrogen will use high pressure ambient temperature hydrogen to drive the turbines. A hot gaseous hydrogen system for the turbines will be developed and used for the hydrogen turbopump tests starting in the third year.

A turbopump bearing and seal test effort will be initiated at the end of the first year to verify seal and bearing performance and life and will be completed in the second year. The assembly of the demonstrator engine turbopump would be completed in the third year.

TABLE 36. NUCLEAR ENGINE TURBOMACHINERY OVERALL TURBOPUMP AND PUMP DATA

	Main	Boost
Turbopump Speed, rpm	75000	19875
Turbopump Shaft Power, hp	2205	144
Turbopump Weight, lb	72	31
Bearing DN, 10^6 mm rpm	1.98	0.196
Pump Flowrate, gpm	1662	1662
Inlet Pressure, psia	107	19
Discharge Pressure, psia	2046	117
Headrise, ft	59254	
Number of Stages	2	1
Stage Specific Speed	1362	
Efficiency	0.81	0.67
Inducer Flow Coefficient	0.15	0.06
Stage Head Coefficient	0.5	0.5
Impeller Tip Speed, ft/sec	1392	456
Inducer Tip Diameter, in.	2.5	5.25
Impeller Tip Diameter, in.	4.25	
Tip Width, in.	0.393	

TABLE 37. NUCLEAR ENGINE TURBOMACHINERY TURBINE DATA

	Main	Boost
Turbine Flowrate	10.95	2.31
Inlet Pressure, psia	1515	1515
Inlet Temperature, °R	1043	1043
Pressure Ratio	1.2	1.2
U/Co	0.488	0.092
Efficiency	0.75	0.23
Number of Stages	1	1
Percent Admission	100	100
Staging	50% R	50% R
"N" Squared Annulus Area, rpm^2, in.^{2}	6.78	0.99
Mean Blade Speed, ft/sec	1500	1500
Mean Diameter, in.	4.58	3.2
First State Rotor		
Height, in.	0.78	0.25

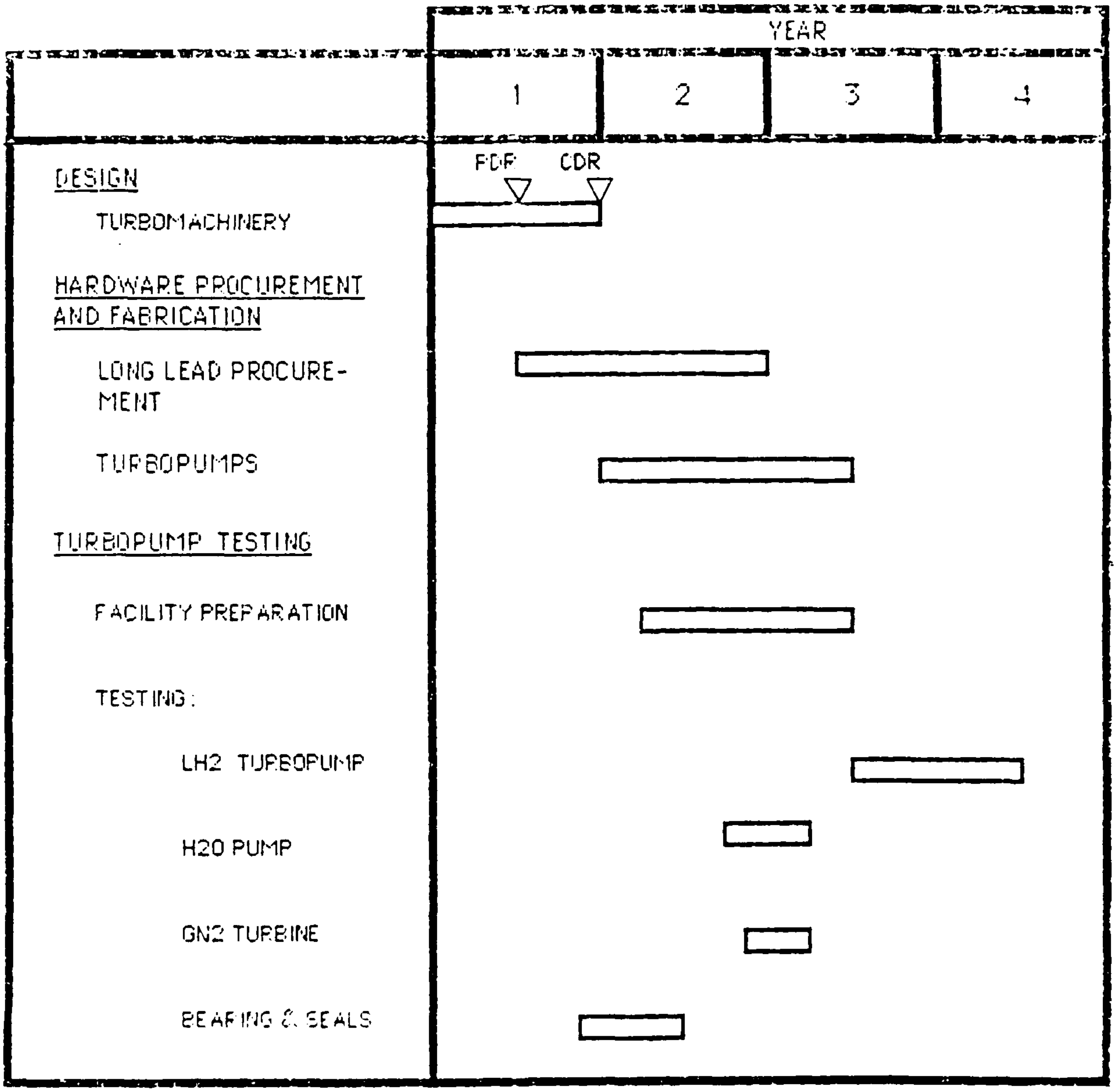


Figure 45. Preliminary turbopump development plan.

Technical Issues

As shown in Table 38, the key nuclear rocket engine turbopump technical issues included the control of the pump operation and pump rotordynamics over the entire throttling range. Due to the potentially wide speed range required due to the 4-to-1 throttling, the shaft critical speeds may overlap the operational range. This potential problem will be resolved by further analysis as the design of the turbopump matures. The turbopump structural issues include low and high cycle fatigue, cavitation damage, and axial thrust balance at throttled conditions.

The occurrence of cavitation during throttling could significantly reduce turbopump operational life and therefore must be evaluated. The variation in the hydrodynamic unbalance between the pump and turbine section over the entire engine operating range must be considered.

TABLE 38. KEY TECHNICAL TURBOPUMP ISSUES

- Pump Control
  - Control over entire throttling range
- Rotordynamics
  - Critical speeds may overlap operating speeds
- Structural
  - Life
  - Axial Thrust Balance

  $\left.\right\}$ Over Range of Operation

4.3.2 Nozzle

Design Description

The preliminary evaluation of the nozzle design resulted in a combination of a regeneratively cooled nozzle with a radiation-cooled nozzle extension. The representative nozzle design conditions chosen are shown in Figure 46. The design chamber pressure was 500 psia with throttling to 25% (125 psia). The nozzle was assumed to be a high area ratio bell nozzle of 80% length. Nozzle area ratios ranging from 300-to-1 to 1200-to-1 are to be investigated.

The regeneratively cooled nozzle section would include the reactor pressure vessel and extend from the subsonic region to a moderate supersonic area ratio and would, in general, consist of a brazed tubular construction. The hydrogen coolant passages would be formed by "booked" tubular cross-sections which would set the local coolant velocity.

The radiation-cooled nozzle extension would be retracted to provide the desired short engine length and deployed in flight for engine operation. The extension would attach to the regeneratively cooled section and extend to the nozzle exit.

To design the final nozzle design, trade studies would be performed to optimize the nozzle area ratio and coolant passage geometry to optimize performance (high specific impulse and low weight) and reduce cost. These trade studies are largely dependent on the mission/vehicle influence coefficient (apayload/aIs). Also, the trade studies would involve the coolant circuit, the detailed tube geometry (including tube splicing), and the thermal environment of the radiation-cooled extension and the regenerative/radiation nozzle joint.

Development Plan

A preliminary nozzle development plan (Figure 47) was configured based on the overall approach Rocketdyne used in the KIWI program. This plan is

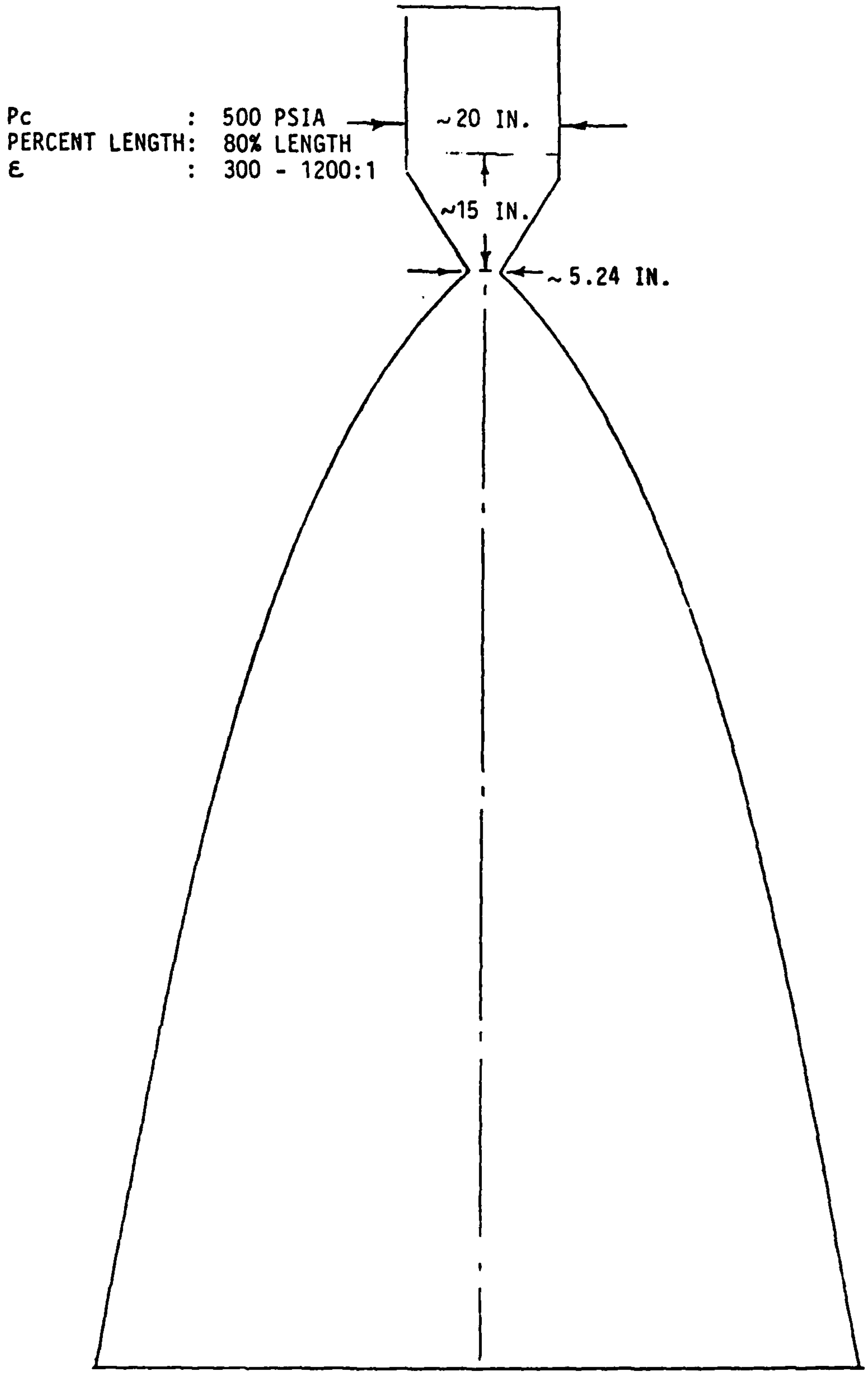
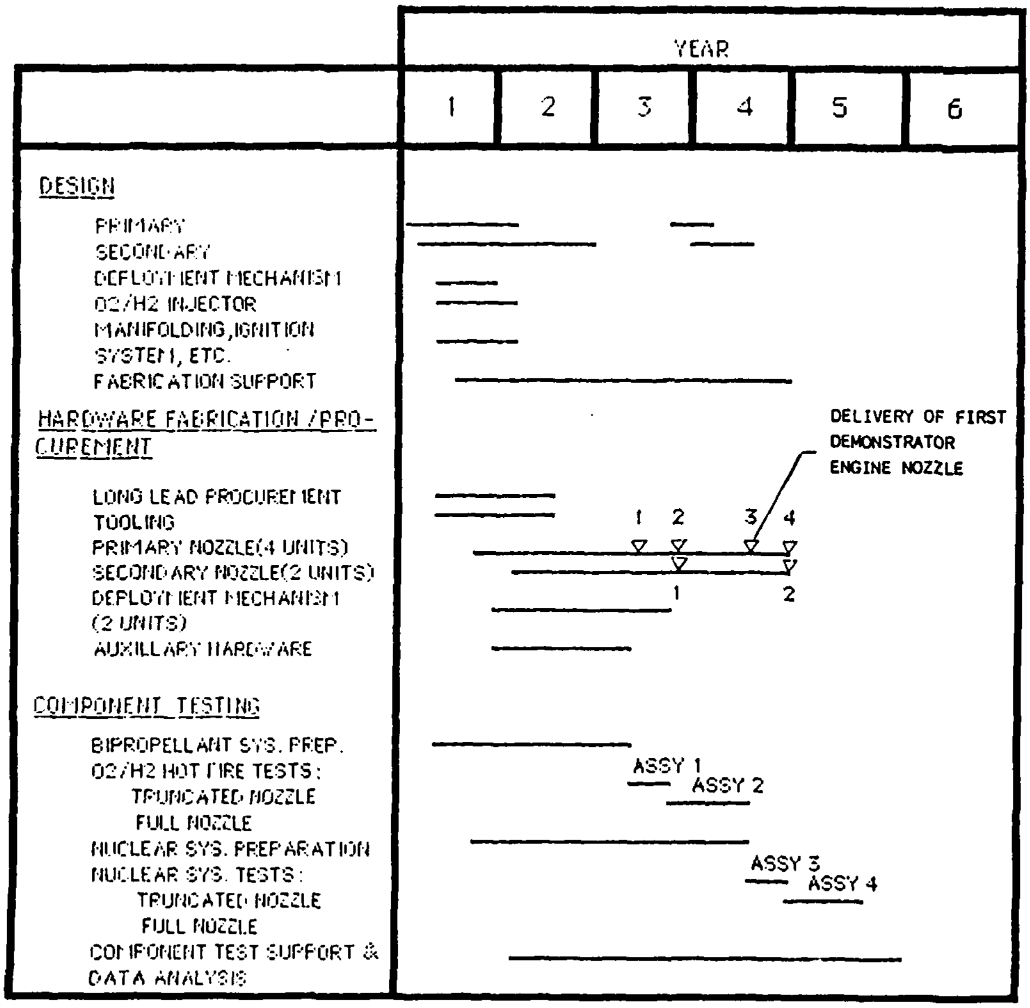


Figure 46. Preliminary nuclear engine nozzle geometry.



ASSEMBLY BUILDS	
ASSY NO.	COMPONENTS
1	NO.1 PRIM. NOZZLE NO.1 O2/H2 INJ.
2	NO.2 PRIM. NOZZLE NO.1 SEC. NOZZLE NO.1 DEPLOY. MECH. NO.2 O2/H2 INJ.
3	NO.3 PRIM. NOZZLE
4	NO.4 PRIM. NOZZLE NO.2 SEC. NOZZLE NO.2 DEPLOY. MECH.

Figure 47. Preliminary nozzle development plan.

a 5-year design, fabrication, and test program with the demonstrator engine nozzle delivered in 3.5 years. Other development plan approaches will be investigated to provide the most efficient and cost-effective approach.

In this preliminary plan, four development nozzles are to be designed and fabricated. Two of these nozzles would be truncated nozzles (only the regeneratively cooled section, without a deployment mechanism. The other two units would consist of both the regeneratively cooled and radiation-cooled extension and would include the deployment mechanisms.

As in the KIWI program, initial tests will be conducted using an O_2/H_2 bipropellant system to simulate the nozzle gas-side thermal environment. These tests would verify structural integrity, cooling characteristics, facility altitude operation and cooling during transients; verify nozzle extension deployment and sealing of the regenerative/radiation extension joint. The reactor/nozzle hardware assembly and tests would follow.

Although the final nozzle development approach will require further evaluation, the overall development plan schedule for the previously described approach is presented in Figure 47. The basic nozzle design is a 2-year effort, with the hardware fabrication and procurement following a half-year later and being completed at the end of the fourth year. The design and fabrication would include the primary nozzle (regeneratively cooled section, the secondary nozzle (the retractable, radiation-cooled extension) the secondary nozzle deployment mechanism, the O_2/H_2 injector, associated manifolds, ignition system, and propellant lines. In the hardware fabrication and procurement phase, the procurement of long lead items would primarily include nozzle tubes and material for manifolds. The tooling would include braze tooling for nozzle tube stack, manifolds, and the secondary nozzle (extension). The nozzle for the demonstrator engine is delivered in 3.5 years.

The testing of the nozzle assemblies will initiate with the O_2/H_2 testing in the third year which would be completed in the fourth year. As shown in Figure 47, the plan also incorporates primary and secondary nozzle

design modifications prior to the procurement, fabrication, and testing of the nozzle with the nuclear reactor. This testing will initiate in the third year and be completed at the end of the fifth year. The test support and test data analysis will continue through the fifth year.

Technical Issues

The key areas of concern regarding the nozzle are presented in Table 39.

TABLE 39. KEY AREAS OF CONCERN - NOZZLE

- Nuclear Radiation Effects on Materials
- Hydrogen Embrittlement of Nozzle Materials
- Cooling of Structure, Flanges, and Bolts
- Cycle Life - 160 Cycles
- Deployment Mechanism and Sealing Between Nozzle Sections
- Use of Nozzle as Radiator

These concerns include concerns associated specifically with the nuclear reactor which involve the radiation effects on the nozzle materials, cooling of structure, flanges, and bolts required as a result of nuclear heating and the utilization of the nozzle as a radiator during reactor shutdown. The concerns also include more conventional concerns such as hydrogen embrittlement influence on nozzle materials, cycle life capability, 4-to-1 throttling ratio, the operation of the deployment mechanism, and the sealing between the two nozzle sections.

As the evaluation and design of the direct nuclear rocket engine progresses, resolution of these concerns will be formulated and, if still unresolved, the test effort will be structured to evaluate and resolve them.

4.3.3 Valves/Control System

Design Description

The control system used for a nuclear rocket must be a blend of conventional rocket engine control system function and a nuclear reactor control system. Preliminary evaluation of the control system requirements for the nuclear rocket resulted in the subsystem functions given in Table 40.

TABLE 40. CONTROLLER OR CONDITION MONITORING RESPONSIBILITIES

Controller	Condition Monitoring
Control Rod Position	Turbopump Turbine Blades
Reactor Mass Flow	Exhaust Plume Gas Chromatograph Turbopump Speeds H_2 Tank Pressures, Temperatures
Control Valve Positions	Pump Vibrations
Decay Heat Management	Valve Pressure Drop Nozzle Pressure Drop Core Fuel Temperatures

The functions can be divided into either controller or condition monitoring responsibilities. The controller and conditioning monitoring functions must be interchanging information continuously for satisfactory engine system performance.

Control system requirements for the nuclear engine are (1) fully throttleable from 25% to 100% power, (2) avoid critical turbopump shaft speeds, (3) avoid cavitation, (4) limit peak fuel temperatures, (5) limit nozzle peak temp/heat fluxes, and (6) allow reduced thrust operation (rather than engine shutdown) when degraded component performance is identified.

The majority of the instruments needed for controller/condition monitoring functions are available from either rocket engine or nuclear reactor vendors. The exception is the exhaust plume gas chromatograph monitor. This instrument will have to be developed for extended use in space and flight-qualified prior to acceptance. Instrumentation designs from nuclear reactor technology are acceptable for nuclear rocket applications; however, lightweight modifications will have to be made and the resulting hardware flight-qualified.

Development Plan

Several control system development tasks are required to provide an engine controller coupled with a condition monitoring system with the combined capability for engine operation over the desired operating envelope. These tasks are identified in sequential order in Figure 48 and presented in a development plan schedule in Figure 49.

Development of rocket engine instrumentation requirements and preliminary selection of available instrumentation suitable for this application will be the first task. In conjunction with this selection, advanced instrumentation requirements will be identified and programs defined to develop these sensors. Both existing and advanced instrumentation will be further developed to meet the weight and reliability requirements for a nuclear rocket.

Control valve requirements specification and preliminary hardware development will be initiated in parallel with the above instrumentation development work. Recent advances in control valve technology for handling both cryogenic and high temperature fluids will be utilized where possible within the environmental constraints imposed by the radiation field from the reactor to define and initiate necessary hardware development. Control valve response, cycle times, and other operating characteristics will be determined for later input to control system simulation studies.

The next series of tasks will utilize the information generated in the above efforts to (1) identify controller and condition monitoring system

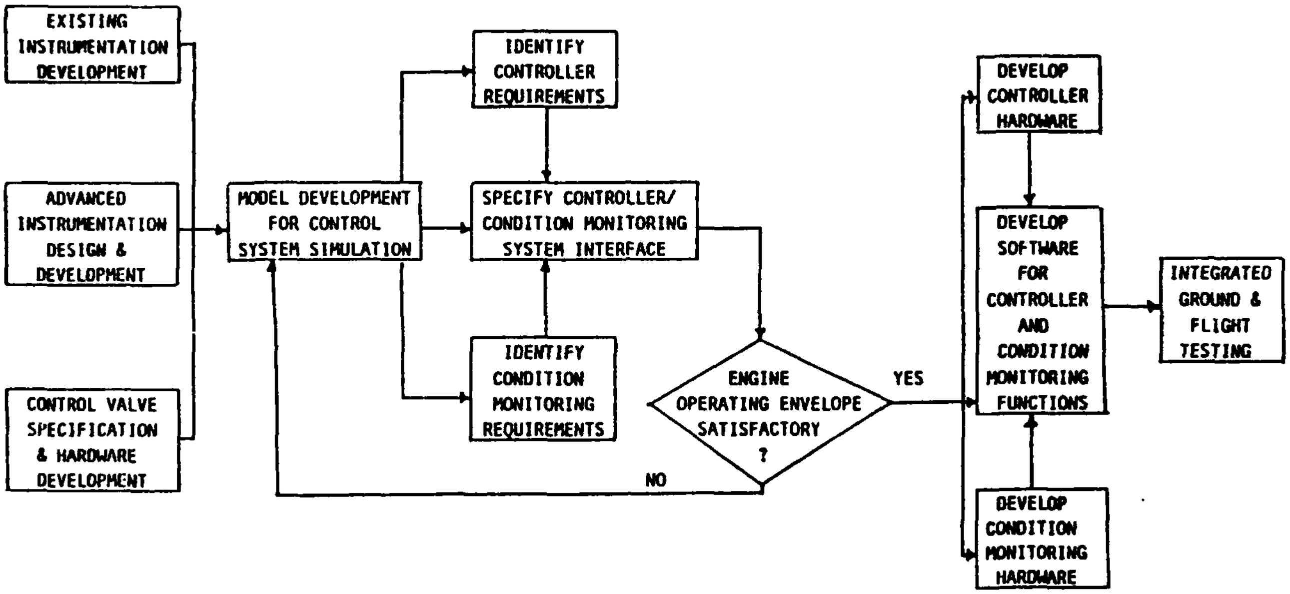


Figure 48. Control system development block diagram.

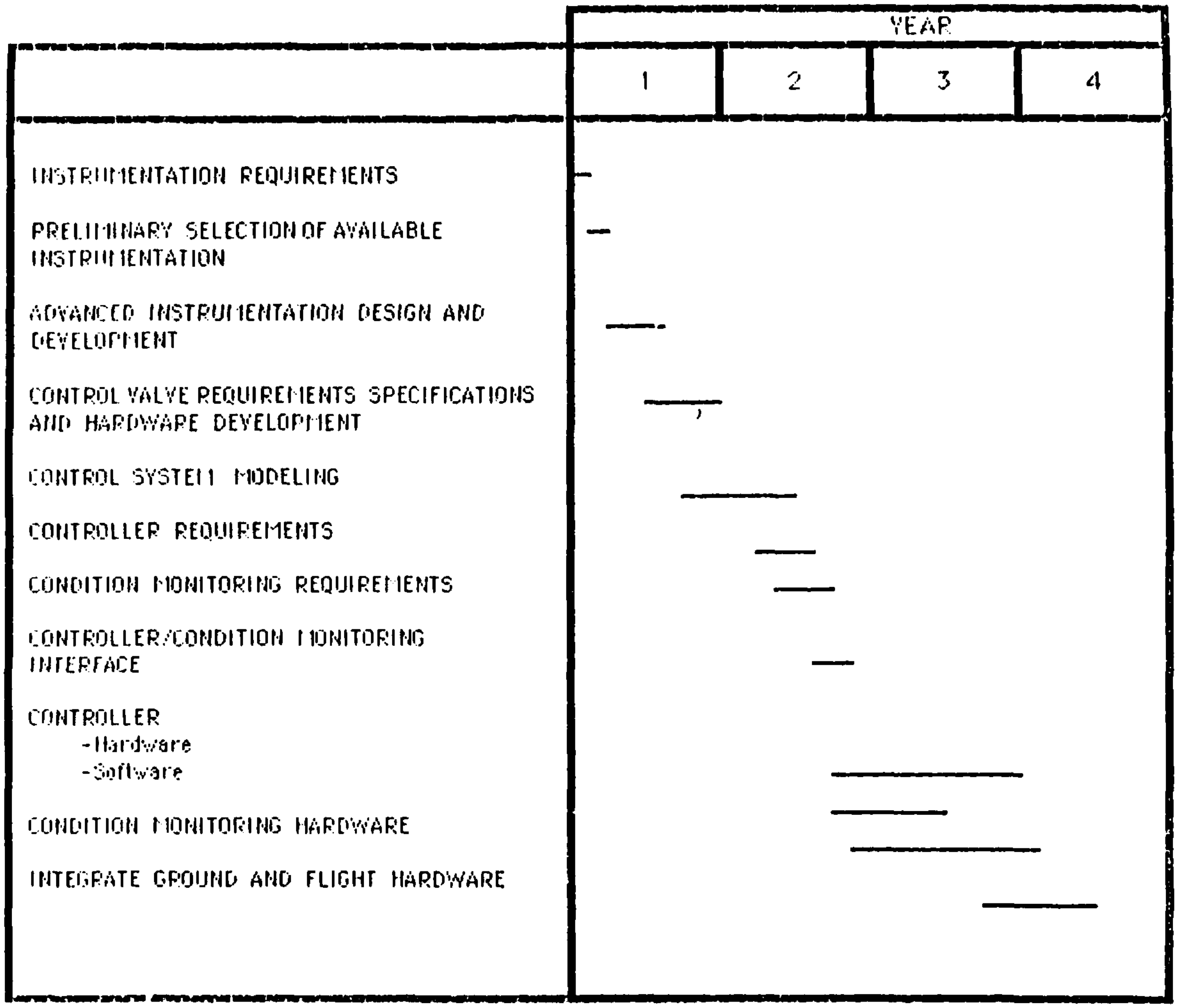


Figure 49. Preliminary control system development plan.

requirements and (2) specify controller/condition monitoring system interfaces required. Instrumentation response times and limits will be combined with control valve operating characteristics and used in a control system modeling effort to simulate overall system response to both normal and off-normal engine operating modes. This system simulation will provide the required controller functions, rates, and interfaces with critical condition monitoring information necessary for both controller and condition monitoring hardware development. Several iterations may be required to test controller, sensor or condition monitoring system changes

and determine the effect on overall engine operation. The modeling effort will grow in sophistication as changes are made to control components, allowing engine operating characteristics to be estimated. This series of tasks is considered complete when sensor controller and condition monitoring system requirements and characteristics combine in the control system simulation model to provide the desired engine operating envelope.

Completion of controller and condition monitoring system requirements will allow initiation of hardware development for both subsystems. These two tasks will be accompanied by a parallel software development task. A mature software package is necessary for both subsystems to function and for a satisfactory continuous interface between the subsystems. Completion of hardware/software development will be followed by both integrated ground and flight testing as indicated in Figures 48 and 49.

Technical Issues

The primary technical issues associated with control system development are: (1) the interface between the "conventional" rocket engine control functions and those additional functions necessary for nuclear reactor control, and (2) the effects of a high radiation environment on both electronic components and valve seat materials. The development plan presented in Figure 48 was selected to provide early attention to and resolution of these issues.

The interface between rocket engine control functions/instrumentation and nuclear reactor control functions/instrumentation will be developed primarily through simulation of engine operation using model development.

Coupled engine response to both normal and off-normal (component failure) operating conditions can be simulated and the required interface between engine and reactor systems identified. Early initiation of this modeling effort (see Figure 48) is necessary to provide sufficient model sophistication and allow for the inevitable iteration between component changes and coupled system response characterization.

Technical issues associated with hardware performance are created by the high radiation field present as part of the normal operating environment. In particular, electronic components and normal valve seat materials (teflon) exhibit considerable sensitivity to both neutron and gamma fluence. Some development work aimed at decreasing sensitivity of electronic components to radiation fields is currently in progress and the results will be beneficial to this program; however, it remains to be determined what the sensitivity can be reduced to.

Control valve seat materials, particularly for cryogenic fluid handling, tend to be teflon-based. Teflon or its derivatives is very sensitive to radiation and critical material properties degrade rapidly. A materials development task early in the overall program is necessary to ensure adequate materials will be available to provide reliable control valve performance in time for engine ground and flight testing. The control valve specification and hardware development task identified in Figure 48 should provide the necessary lead time.

4.4 Cost Estimates

Estimates of component costs were developed for the non-nuclear engine components. These values have developed parametrically using a combination of cost estimating techniques. Due to the uncertainty in final design configuration, these values are ROM only. Table 41 provides the rationale for the component cost estimates. Table 42 contains the low, medium, and high cost estimates for these components.

TABLE 41. RATIONALE FOR NUCLEAR ENGINE DEVELOPMENT COST ESTIMATES

Cost Element	Cost Estimate	Rationale
o Turbopumps		
- Hardware	Medium	From Rocketdyne's Parametric Rocket Engine Component Hardware Cost Estimating Relationships (CERs), based on turbopump weight and complexity. The CERs are anchored in historical data of turbopumps.
	Low, Hi	CERs are generally assumed to be accurate within ±20%; therefore, the low and high values are 80% and 120% of the "medium" value.
- Engineering, Test, Mgmt	Medium	From Rocketdyne's Parametric Rocket Engine Non-Hardware CERs, based on rocket engine thrust and turbopump hardware unit cost.
	Low	From the parametric cost model, price (by RCA), using turbopump weight, complexity, platform and development time factors.
	High	Twice the "Medium" estimate due to presently unknown configuration requirements (e.g., throttling) which may lead to significantly increased design and testing effort.
o Nozzles		
- Hardware	Medium	Regeneratively Cooled Part: From Rocketdyne's CER for tubular thrust chambers/nozzle based on nozzle weight for r=600 Radiation-Cooled Part: Estimated at 50% of regeneratively cooled nozzle based on wall temperature of 2500°R and expansion ratio of r=600 Deployment Mechanism: Based on factoring obtained from Rocketdyne's cost estimating of DTV engines.
	Low	80% of the medium value, based on same rationale as described under "Turbopump Hardware".
	High	ROM estimate by combustion devices group, taking into account potentially more hardware required to simulate high nozzle wall temperatures and long life; i.e., bipropellant injector, manifolds and chemical propellant combustion chamber.
- Engineering, Test, Mgmt	Medium	From price model using total nozzle weight, complexity. Platform and development time factors.

TABLE 41. RATIONALE FOR NUCLEAR ENGINE DEVELOPMENT COST ESTIMATES (CONCLUDED)

Cost Element	Cost Estimate	Rationale
	Low	Success-oriented design and test program, based on past experience.
	High	50% higher than "Medium" value to take into account higher amount of development work due to bipropellant testing and investigation of flow separation at very high expansion ratio (r=600-1200), if these are required for high performance.
o Engine Control System		
- Hardware	Medium	Based on the cost of two simplified SSME Block II controllers. Controllers assumed if deep, continuous throttling and engine Health Monitoring System (HMS) is required.
	Low, Hi	"Medium" value û50% to account for presently unknown control hardware required, depending on control philosophy (throttling?) and HMS approach.
- Engineering, Test, Mgmt	Medium	Engineering estimate based on prior OTV controller work for an expander cycle without combustor and with monopropellant.
	Low	50% of "Medium" value based on the assumption that only a sequencer is required for step-throttling, and a simple HMS which is an annex to the vehicle HMS.
	High	Twice the "Medium" value to account for a more complex control system, a more sophisticated HMS, and potentially significant integration with the reaction control.
o Engine System Integration & Test Support For 5 Years by Contractor	Medium	Based on a sustained engineering and test support effort at the test bed site using 20 equivalent persons (EPS) for 5 years. Includes EP salary, wrap factors and computer time, but no additional material or hardware than specified in "Ground Rules".
	Low	50% of "Medium" value for success-oriented test bed program with larger participation of government personnel.
	High	Twice the "Medium" value to account for increased contractor support due to potential integration problems.

TABLE 42. NUCLEAR ENGINE DEVELOPMENT COST ESTIMATING SUMMARY (M$. 1987)

	Low	Medium	High
Turbopumps			
Development Hardware	2.0	2.4	3.7
Engineering, Testing, Mgmt	6.6	16.0	32.0
	8.6	18.4	35.7
Nozzles			
Development Hardware	10.0	12.4	18.0
Engineering, Testing, Mgmt	17.0	41.1	52.0
	27.0	53.5	70.0
Engine Control System			
Development Hardware	3.0	6.0	9.0
Engineering, Testing, Mgmt	10.0	20.0	40.0
	13.0	26.0	49.0
Engine System Integration & Test Support for 5 Years	6.3	12.6	25.0
Total	54.9	110.5	179.7

GLOSSARY

AB	Aerobrake
A&C/O	Assembly and Checkout
AFAL	Air Force Astronautics Laboratory
ANRE	Advanced Nuclear Rocket Engine
ASE	Auxiliary Support Equipment
CASTLE	Cycling Astronautical Spaceships for Transplanetary Long-Duration Excursion
CCV	Cooldown Control Valve
C&DH	Communications & Data Handling
CPF	Cost Per Flight
CTE	Coefficient of Thermal Expansion
CVD	Chemical Vapor Deposition
DDT&E	Design, Development,Test, and Evaluation
DOD	Department of Defense
DOE	Department of Energy
EP	Electric Propulsion
EPS	Electric Propulsion System
ETR	Eastern Test Range
EVA	Extra-Vehicular Activity
FMA	Failure Mode Analysis
FSAR	Final Safety Analysis Report
GB	Ground Based
GEO	Geosynchronous Earth Orbit
GN&C	Guidance Navigation & Control
GSE	Ground Support Equipment
INEL	Idaho National Engineering Laboratory
INSRP	Interagency Nuclear Safety Review Panel
Isp	Specific Impulse
IVA	Inter Vehicular Activity
LCC	Life Cycle Cost
LCV	Large Cargo Vehicle
LDAV	Lunar Descent/Ascent Vehicle
LEO	Low Earth Orbit
LEP	Low Earth Platform
LERC	Lewis Research Center
LLO	Low Lunar Orbit
LLOX	Lunar Liquid Oxygen
LMO	Low Mars Orbit
LOX	Liquid Oxygen
LTS	Lower Thrust Structure
MDAV	Mars Descent/Ascent Vehicle
MEO	Medium Earth Orbit
MLI	Multi-Layer Insulation
MPD	Magneto Plasma Dynamic
MSS	Mars Space Station
NASA	National Aeronautics and Space Administration
NCV	Nozzle Control Valve
NEP	Nuclear Electric Propulsion
NERVA	Nuclear Engine for Rocket Vehicle Application
NPS	Nuclear Power System

OMV	Orbital Maneuvering Vehicle
O&S	Operations & Support
OST	Office of Science and Technology
OTM	Orbiting Trim Maneuver
OTV	Orbit Transfer Vehicle
PBR	Particle Bed Reactor
PL or P/L	Payload
PSAR	Preliminary Safety Analysis Report
PSOV	Propellant Shutoff Valve
RCS	Reaction Control System
RFB	Refurbish
ROM	Rough Order of Magnitude
SAIC	Science Applications International Corporation
SB	Space Based
SDI	Strategic Defense Initiative
SER	Safety Evaluation Report
SSE	Station Support Equipment
STAS	Space Transportation Architecture Study
STS	Space Transportation System
TBCV	Turbine Bypass Control Valve
TCM	Trajectory Correction Maneuver
TDBV	Turbine Discharge Block Valve
USAR	Updated Safety Analysis Report
UTS	Upper Thrust Structure
W/AB	With Aerobrake
WO/AB	Without Aerobrake
WTR	Western Test Range

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APPENDIX A

BASELINE ENGINE DESCRIPTION

APPENDIX A
BASELINE ENGINE DESCRIPTION

The baseline engine characteristics used in the evaluation of life cycle costs were based on the technology and experience developed in the NERVA (Nuclear Engine-Rocket Vehicle Application) programs.[1,2] This program, cosponsored by the AEC (now DOE) and NASA, was initiated in 1960 and terminated, for lack of a defined mission, in 1972. During that period, some $1.4 billion of then-year dollars were spent in the successful development of a technology suitable for space nuclear propulsion, including the ground testing of over 20 reactor systems (including the fuel, components, advanced materials, and methodology for the analysis of safety, reliability, and operability that are as valid today as when they were first developed). The last major activity performed during the NERVA program, prior to project termination, was the integration of all of the technology into the preliminary design of the first flight-rated reactor system. It successfully passed the equivalent of an Air Force Preliminary Design Review (PDR).[3] Figure A-1 shows a mockup of the NERVA engine.

The paragraphs below summarize the specific experience and technology status of the NERVA program as it applies to a nuclear engine that could be designed, built, and demonstrated in a program with minimal performance, schedule, or cost risk.

A.1 PERFORMANCE

The nuclear reactor performance capabilities for the NERVA - Derivative are derived from the conclusions based on 20 reactor tests with over 1000 min of integrated operating time.[4] During these tests, 60 min of continuous rated power was demonstrated in the 1100 MWt NRX-A6 (experimental reactor), as well as 28 full startup and shutdown cycles of a fully integrated nuclear engine system in the XE-Prime engine tests.[5] Flexibility of operation and control of the engine were demonstrated throughout its full operating map. The smaller (500 MWt) Pewee 1 reactor subsystem testing[6] demonstrated the principal hardware and performance

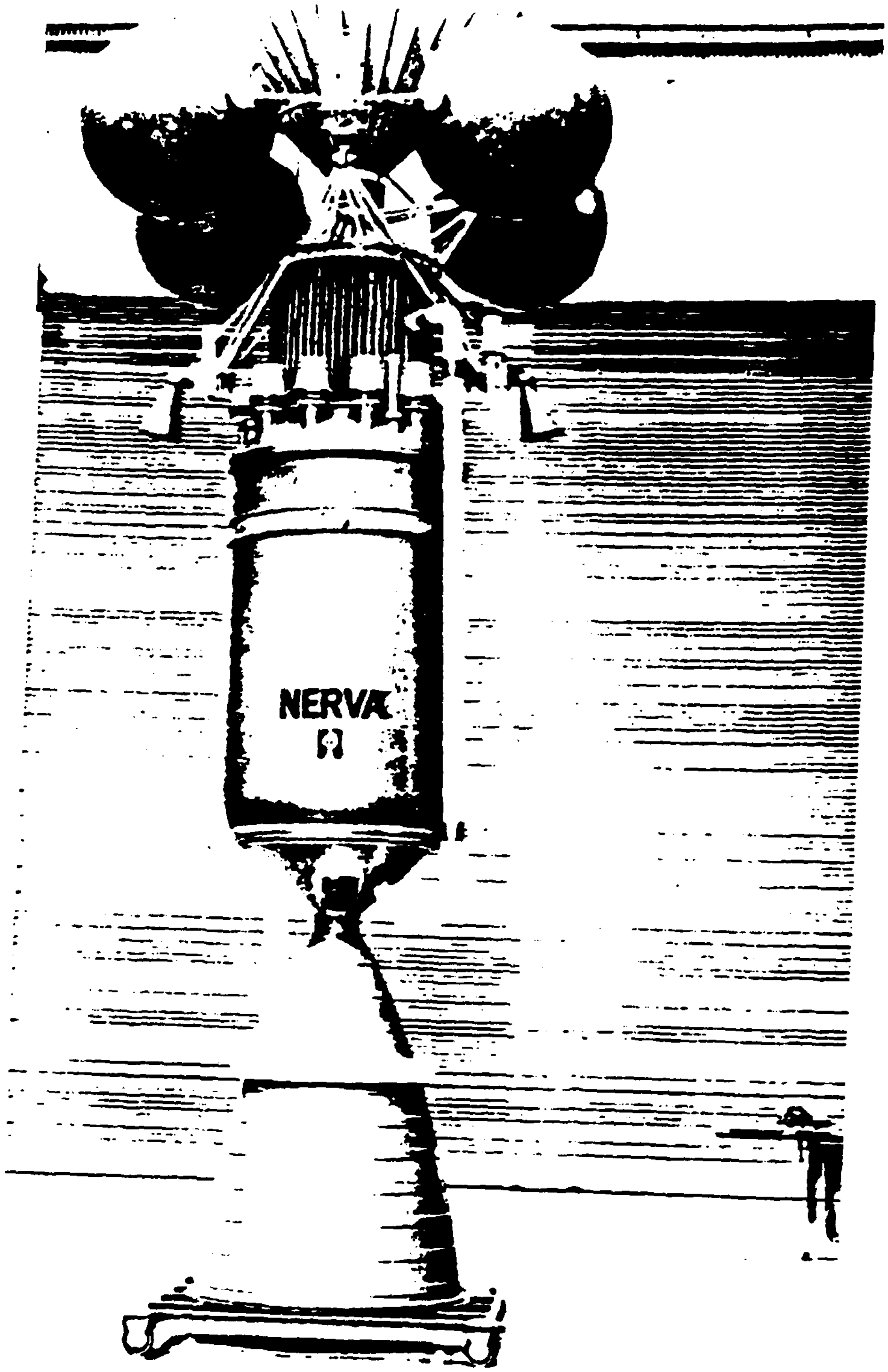

Figure A-1. Full scale mockup of NR-1 flight engine rated at 1500 MW_t and 75,000-lb thrust.

characteristics for the small NERVA - Derivative reactor, which has a nominal 300 MWt power level.

The operational versatility of the NERVA engine is indicated by the operating map shown in Figure A-2. As indicated by the upper horizontal line of the operating map, the engine can be throttled to 40% of full power while retaining full power Isp. The figure also includes the results of engine test data that show the controllability of the engine during startup and shutdown.

NERVA XE Engine Test
EP-V-C 11 JUNE 69

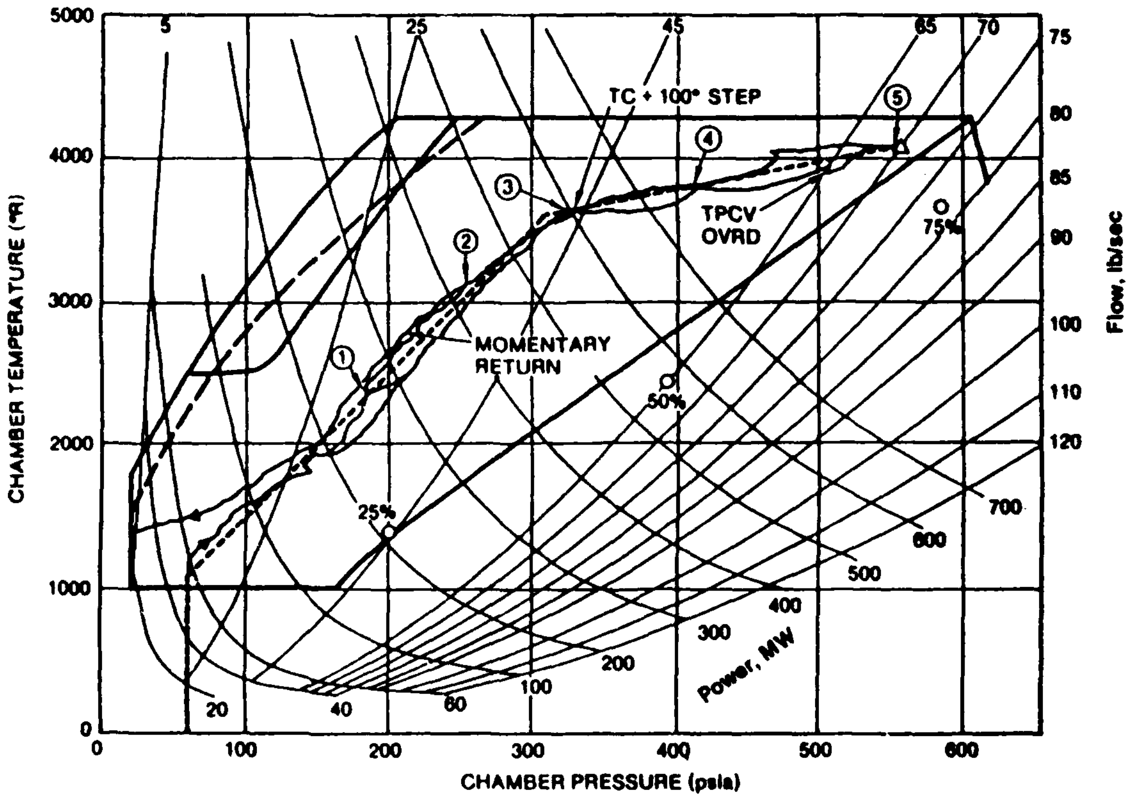

Figure A-2. Typical NERVA engine operating map.

Modifications considered for the current small engine design include the incorporation of the NRX A6 type lateral support system, which demonstrated its capability to handle extensive vibration and shock loads in both out-of-pile and NRX-A6 reactor tests. The use of a higher temperature and more reliable fuel based on the composite fuel technology was successfully demonstrated in the Nuclear Furnace[7], and subsequent fuels with improved coatings were demonstrated in an electrically heated simulation. These fuels were tested for 10 hr and 60 cycles at temperatures and power densities representative of design conditions for the reference engine used in this study.[8,9] Pure carbide (UC.ZrC) fuel elements are in an early stage of development and show promise to attain reactor operating temperatures in excess of 3100 K, with a resulting specific impulse greater than 970 s. The reactor design for these new fuel elements is similar to that for the composite fuel elements, but is modified to accommodate the lower thermal-stress tolerance of the carbide elements.[8]

A.2 SAFETY

Safety was a critical issue during the NERVA program as it is today. The analyses, design, and testing were all done within the restraints of the nuclear safety requirements that existed at the time.[10] Nuclear safety considerations included the design for avoidance of inadvertent criticality during potential launch or land operations, analysis of reentry conditions, and the safety of assembly and transport of fully loaded reactors from the manufacturing site in Pittsburgh to the test site at Jackass Flats, Nevada. Twenty reactors were operated during the program. The results obtained from the operation of these test articles have provided a valid design basis for the small engine used in the current study.

A.3 RELIABILITY

Since one of the major thrusts of the NERVA program was the utilization of the nuclear engine in a manned mission, reliability was a key issue throughout the program.[3,10] Very formal failure mode and

effect analyses were performed to identify potential problem areas and avert them. This was begun during design and continued through assembly and test operations to assess potential operational effects, with proof testing and/or redesign done to achieve the desired reliability objectives.[11] The program included not only the testing of full size reactors, but also the testing of over 20,000 fuel elements, and the manufacture and operation of all subsystem components, control devices, instrumentation, and other non-nuclear and servicing components of the reactor system. Proof testing of the system reliability took place in the major full nuclear engine tests, in particular the NRX-A1 through A6 (which demonstrated continuous 60-minute full power operation) and the XE-Prime which underwent 2b full operational startups. As a result of the design, analyses, and test activities accomplished during the NERVA program, a reasonable, although not complete, data base has been established to be able to project the reliability of the small nuclear engine with a good degree of confidence. Westinghouse has retained all design documentation for the NERVA concept engine. Experienced personnel are still available and engine fabrication and testing could be rapidly restarted.

A.4 COST AND MASS

The definition of engineering design and purchase specifications for the NERVA reactor and engine components provided a basis for developing and qualifying acceptable vendors for the hardware and subsystem procurements during the NERVA development program. Based on this experience, the fabrication effort, schedule and costs for the test articles, as well as mass, could be well established.[12] From this data base, scaling where required for size and number of pieces (such as fuel elements and control drums), a reasonable mass and cost estimate could be derived for a reference reactor. The costs were escalated from the then-year costs to today's cost for the mission studies. Quality assurance and spare part considerations were included, but no cost increments for engineering/support or development of vendors or tooling were included, nor did these costing data consider improved materials, fabrication, and manufacturing capabilities. Based on these considerations it is estimated the reference engine used in this study could be produced for 30 million

dollars. Table A-1 shows the mass estimates for the baseline ANRE compared to those for the SNRE γ engine.

TABLE A-1. "SMALL ENGINE" MASS ESTIMATES (kg)

Component	SNRE Engine	ANRE NERVA Derivative
Reactor Core and Hardware	868	868
Reflector and Hardware	569	469
Shield	239	186
Pressure Vessel	150	40
Turbopump	41	41
Nozzle and Skirt Assembly	224	224
Propellant Lines	15	15
Thrust Structure and Gimbal	28	28
Valves and Actuators	207	107
Instrumentation and Electronics	159	59
Contingency	50	50
Total	2550	2087

NOTE: The above SNRE total includes 100 kg for instrumentation cabling and electronics that are mounted on the stage rather than on the engine and 36 kg for the thrust vector actuators.

The original SNRE γ engine design was based on component weights and performance data derived during NERVA program development. Since the termination of this program, non-nuclear engine component development has continued for chemical rocket engine applications. The baseline design used in the mission analysis contains primarily components from the SNRE engine with low risk updated values for some component weights, as shown in Table A-1.

The reactor and associated hardware weights remain unchanged from the original design; however, the reflector/hardware assembly has been decreased 100 kg to allow for newer lightweight control drum drive motor assemblies. The reactor shield weight has been lowered 53 kg to account for tungsten/lithium-hydride composite shielding advances in thermonuclear fusion research and the pressure vessel mass has been reduced to that

required for a titanium alloy vessel. The turbopump and nozzle assembly weights remain the same; however, the actual components will be significantly different. The engine turbopump specifications were 874 psid, 18.7 lbm/sec, and 47,000 rpm. A turbopump built today using these volumes would weight about 23 kg (versus the 41 kg for the engine). However, a 41 kg turbopump will provide for (a) a higher pressure necessary to regeneratively cool a larger nozzle and (b) a lower speed to increase pump reliability/lifetime.

The nozzle/skirt assembly mass also remains unchanged but employs significantly advanced technology. The original engine used regenerative cooling out to an area ratio of 25:1 and an uncooled skirt extension out to 100:1. Using today's technology for an engine operating solely in a vacuum, regenerative cooling to an area ratio of 300:1 and an uncooled skirt extension to 600-900:1 are not unreasonable and would weigh about the same as the original nozzle. The higher area ratio will maximize delivered Isp for a given chamber condition.

The other major component where low risk weight savings is easily attainable was in the control valves. The original valves were pneumatic with aluminum valve bodies and inconel gates. Recent R&D activities have dropped total control valve mass for a 1.5-in. ID LOX valve/actuator to 6 lbm. These advances are reflected in the 100-kg savings in mass projected for the NERVA derivative engine control valves.

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APPENDIX B

GROUND RULES AND ASSUMPTIONS, LIFE CYCLE COST (LCC) ANALYSIS OF CHEMICAL/NUCLEAR STAGES

APPENDIX B

GROUND RULES AND ASSUMPTIONS, LIFE CYCLE COST (LCC) ANALYSIS OF CHEMICAL/NUCLEAR STAGES

LCC analysis for comparative costing of chemical and nuclear stages was done using conceptual level models. Historical program costs were applied to structure a baseline methodology. The following groundrules and assumptions were used to develop estimates for (a) the chemical SB OMV/OTV and (b) nuclear SB OMV/OTV stages. (References for ground rules are based on the following Martin Marietta contracts with NASA: OMV-NAS-8-36115; OTV-NAS-8-36108; STAS-NAS-8-36618.)

Rules Common to Chemical and Nuclear Stages

- All costs are in constant 1986 dollars and do not include fees and contingencies.
- No learning curve was applied due to small production quantities of SB OMV/OTV stages.
- Cost for initial OMV/OTV delivery to orbit: $70 million per launch event.
- Launch vehicle capability: 150 K-lb to LEO; 109 K-lb to space platforms.
- Launch vehicle cargo envelope: 25 ft (diam) and 90 ft (length).
- Launch costs include delivery of initial stages, platforms, and spares.
- Mission operations costs are based on a fixed 35 manyear level of effort.
- Payload transportation costs are assessed per the STS Reimbursement Guide, e.g., charge algorithms, platform cost.

Rules Common to Space Stations and Nuclear Stage Platforms

- Propellant delivery costs as per groundrules of the NASA OTV Program (67% at "space available" rates; 33% at dedicated tanker rates.)
- Refurbishment costs: spread equally over all missions.
- Platform and Space Station use charges: 250 K per payload.
- IVA servicing charge: 18 K per robotic/crewman hour.
- A minimum of two operational stages will be docked at space platforms at all times.
- Platforms will be able to support the number of missions scheduled.

Rules for Chemical Stages only:

- Hardware costs for Integral Stages are not included, but propellant costs for delivery to parking orbits are.
- SB OMV/UTV engine service life: 10 missions
- SB OMV/OTV tankage replacement:[a] 40 missions
- SB OMV/OTV airframe, avionics, etc., service life: 40 missions
- SB OTV aerobrake service life: 5 missions

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a. Service life data based on our experience with the NASA OTV program.

Rules for Nuclear Stages only:

- o The baseline nuclear engine uses technology proven in the NERVA program.
- o ANRE service life: 80 missions
- o Service life of airframe, structure, avionics, etc: 100 missions

APPENDIX C

SAFETY POLICY, GUIDELINES, AND REVIEW PROCESS

APPENDIX C

SAFETY POLICY, GUIDELINES, AND REVIEW PROCESS

United States policy on the use of nuclear reactors in space requires that stringent design and operational measures be used by the U.S. to minimize potential interaction of radioactive materials with the populace and the environment and to keep exposure levels within limits established by international standards. The U.S. policy has been presented in a number of papers to the United Nations Scientific and Technical Subcommittee on the Peaceful Uses of Outer Space[1,2] and in the U.S. concurrence to the reports issued by that subcommittee.[3,4,5]

The U.N. Working Group believes that the bases for a decision on a nuclear power source should be technical provided that exposure risk is maintained at an acceptably low level. The Working Group defines that level by recommending that the annual dose equivalent limit for workers be set at 50 mSv (5 rem) whole body dose (or equivalent doses to parts of the body). Furthermore, an annual dose equivalent limit for the most highly exposed members of the public (the critical group) of 5 mSv (0.5 rem) from all man-made sources should not be exceeded during the normal phases of a nuclear power system mission. The Group has not yet set specific guidelines for accident conditions.

C.1 GUIDELINES FOR NUCLEAR SAFETY IN SPACE

U.S. safety guidelines are delineated in DOE criteria[6] and the current space nuclear power program, SP-100, specifications. These safety criteria and specifications require that credible launch pad, ascent, abort, or reentry accidents resulting in Earth impact not result in a sustained nuclear fissioning source. The radiation from reentered reactor material, whether scattered by an explosion or imparting intact, must be well within national and international safety standards. The reactor is also required to have at least two independent systems to ensure shutdown. An orbital altitude boost system is to be provided by the mission agency (for short-lived orbit missions) to boost the reactor into high orbits for

radioactivity decay following mission completion or upon mission failure. These policies are considered adequate under current circumstances.

Current guidelines are also given in JSC 30307, "Nuclear Safety Guidelines for Space Applications" with a current update being proposed in BB99231. These guidelines aid in the elimination and/or control of nuclear related hazards by addressing nuclear system design, nuclear support system design, operations during flight, and operations during ground activities. Hazards, defined as potential risks in a system, are categorized as collision, contamination, corrosion, electrical shock, explosion, fire, injury and illness, radiation exposure, and radiation and temperature extremes.

Ground personnel and general population radiation exposure limits are defined in Title 10, Part 20, of the Code of Federal Regulations. A 4 km diameter, controlled exclusion area around the launch pad is called for during prelaunch and launch activities. Launches containing radioactive materials are to be conducted with the prevailing winds blowing away from populated areas. Provisions for detection and decontamination must be made at landing sites in the event a nuclear source is on-board and in the event of radiation leaks. Flight termination impact areas for nuclear hardware outside the continental shelf, preferable in deep ocean areas, are to be investigated to minimize hazards to the ecology and general populace. Safety and destruct systems are to be considered to reduce the potential for earth impact and release of radioactive material on the Eurasian continent. Radioactive payloads must be able to:

1. Withstand the worst-case pressure gradient associated with the most credible scenario for detonation of the liquid and/or solid rocket propellant on the launch pad.

2. Withstand the worst-case temperatures created by the most credible source of fire associated with the detonation and burning of the liquid and/or solid rocket propellant.

3. Withstand reentry from Earth orbit and impact on land or water with a reentry trajectory that will generate the highest credible mechanical shock and vibration.

4. Withstand worst-case credible combination of pressure gradients, temperature, and vibration associated with detonation of the launch vehicle at any time during the launch and ascent phase.

The following are also required:

1. A positive and permanent shutdown system for malfunctioning reactors and for reactors which have completed their missions.

2. A redundant, automatic means of reactor shutdown to control operation under all contingencies.

An important, proposed provision is that permanent disposal be in a solar orbit of at least 0.84 of the Earth's orbital radius.

C.2 U.S. AEROSPACE NUCLEAR SAFETY REVIEW PROCESS

Every U.S. nuclear-fueled supply that is considered for use in space must undergo a safety review process. This process establishes that the potential risks associated with the nuclear energy source use are commensurate with the anticipated mission benefits. A formalized review process has been developed for evaluating the safety aspects of nuclear system launches. At the center of this process is the Interagency Nuclear Safety Review Panel (INSRP) comprised of representatives from the Department of Energy (DOE), the National Aeronautics and Space Administration (NASA), and the Department of Defense (DOD). These agencies are responsible for evaluating mission safety for each launch. DOD and NASA personnel are involved because these two government agencies have safety responsibilities and expertise, both as launching organizations and as user organizations of space nuclear power. DOE has statutory responsibility for the safety of space nuclear systems.

The evaluation process consists of the following elements:

1. The lead or sponsoring agency directs the manufacturer of the nuclear system to write a Preliminary Safety Analysis Report (PSAR) or Updated Safety Analysis Report (USAR) describing all aspects of mission safety.

2. Safety Analysis Reports are distributed to the members of the INSRP and each member agency conducts its own review and critique of the PSAR or USAR.

3. A meeting of the INSRP is held with member agencies and their mission hardware contractors (launch vehicle, nuclear fuel, power system, space vehicle, etc.) in attendance. The results of the independent reviews are presented and discussed at this meeting. Action items are generated to resolve any open questions or issues.

4. The nuclear system contractor, with input from other agencies/contractors responsible for action items, writes a Final Safety Analysis Report (FSAR) taking into account the PSAR or USAR critiques and any appropriate new information.

5. Elements 2 and 3 are repeated with the FSAR.

6. The INSRP generates a Safety Evaluation Report (SER) that accompanies the request for Presidential approval of the launch.

The SER is a risk assessment made by the INSRP. In addition to the information provided in the FSAR, the SER also contains analyses and tests performed by many technical people from government agencies, laboratories, and universities. The SER evaluates potential human exposures to radiation and the probability of exposure during all phases of the mission. The INSRP submits the SER to the heads of DOD, NASA, and DOE for their review with the INSRP recommendations/conclusions about the safety of the proposed mission. The key concept here is that the INSRP makes recommendations but

does not make any final decision. The head of the agency that wants to fly the nuclear system then must request launch approval from the President through the Office of Science and Technology (OST). The heads of the other two agencies represented on the INSRP may choose to support the user agency with statements of support. The OST will review the user agency requirements and may send the request to the National Security Council for review. The ultimate authority for launch and use of the NPS lies with the President of the United States. Figure C-1 illustrates the review and approval process, which requires presidential approval for the launch and use of nuclear systems in space.

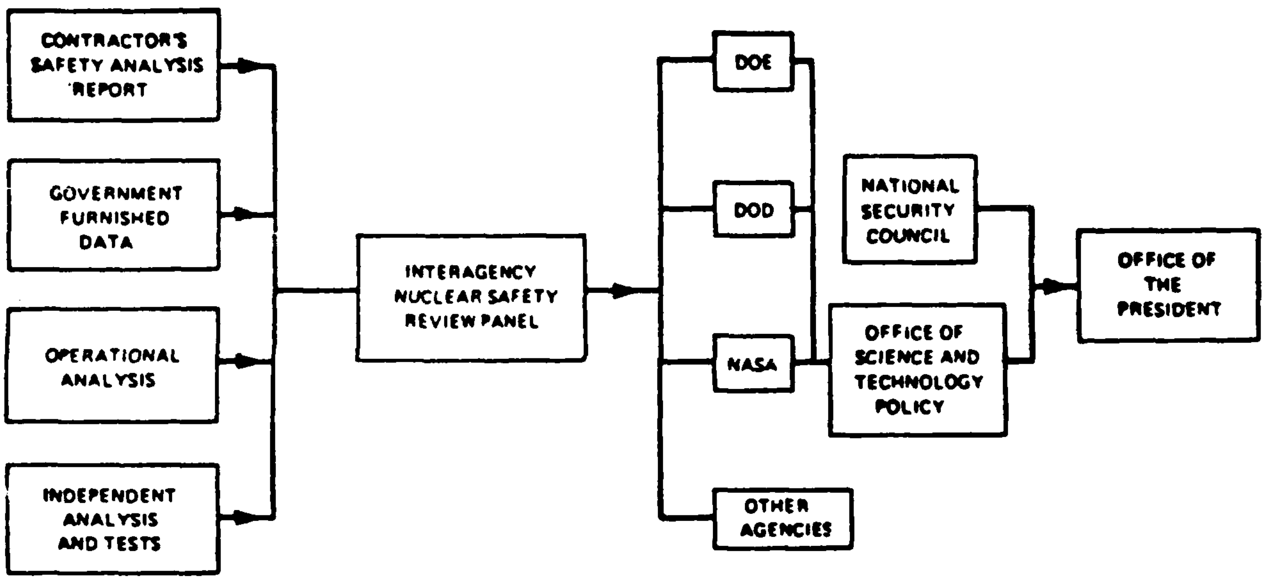

Figure C-1. Safety review and launch approval process.

Because safety features are designed into U.S. space nuclear systems from the very beginning, this safety review process is actually an integral part of the overall flight system development and in no way constrains the overall mission schedule.

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APPENDIX D

SAIC LETTER RE INDEPENDENT REVIEW OF MISSION ANALYSES

SAIC
Science Applications International Corporation

February 26, 1987

Mr. Jack Ramsthaler
EG & G
1580 Sawtelle
Idaho Falls, Idaho 83415

Dear Mr. Ramsthaler:

This letter report addresses the assessment performed by SAIC, per your request, in review of Martin Marietta's work on mission applications of advanced nuclear thermal rockets. Our analysis was based on an examination of presentation material provided by Martin, some independent calculations of payload performance, and attendance at a briefing by Martin during which there was ample opportunity to ask questions and offer critical suggestions. Although our assessment analysis was necessarily brief given the time available, I believe it is sufficiently thorough in a preliminary sense.

Basically, our independent audit of payload delivery performance showed substantial agreement with Martin's data - cetainly within 10 percent as a worst case. Our trajectory simulations included finite-thrust gravity losses which was ignored by Martin; however, these losses amounted to less than 3 percent for a small nuclear rocket stage (13,250 lbs propellant load), or less than 11 percent for a large stage (36,900 lbs propellant load). On the other hand, Martin correctly accounted for engine shutdown/cooling effects in their analysis, which we did not.

Perhaps the most important aspect of our review was the suggestion that the cost benefit comparison with cryogenic OTV's should be presented in parametric form showing clearly the effect of assumed launch cost. You will recall that the operations elements of the life-cycle cost breakdown was a dominant fraction of total cost in the baseline case, and that this element depended

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strongly on assumed launch costs. Reduced launch cost anticipated for future operations may still show a significant benefit for nuclear OTV's, but this needs to be tested.

Finally, based on recent work on manned Mars missions, I might point out that nuclear rocket application offers very substantial reduction in the total weight that must be assembled in Earth orbit, or equivalently, in the number of heavy lift vehicle (HLV) launches that are required. With the SNRE as a precursor development to large-sized nuclear thermal engines, the future exploration of Mars in a sustained outpost scenario might well be benefitted by the availability of this advanced propulsion technology.

I hope that this letter report is sufficient for your purpose. Please let me know if there is any further information that I can provide. SAIC would be pleased to participate in any extended analysis and assessment of nuclear rocket mission applications, particularly as it relates to advanced planetary exploration (automated and piloted) and utilization of lunar resources.

Sincerely,

SCIENCE APPLICATIONS INTERNATIONAL CORPORATION

Alan L. Friedlander

Alan L. Friedlander
Project Manager and Senior Analyst
Space Sciences Department

ALF:dkl

cc: Dave Buden, SAIC/Albuquerque

APPENDIX E
ELECTRIC PROPULSION SYSTEMS

APPENDIX E
ELECTRIC PROPULSION SYSTEMS

As noted in the Introduction, this report includes comparisons of the potential competitiveness of nuclear rockets to electric propulsion systems. A detailed analysis of the cost/benefits of electric propulsion systems is well beyond the scope of this report, although it is appropriate to make a qualitative assessment based on information available in the general literature.

This appendix provides a brief development status review, discusses some of the unique characteristics affecting applications of electric thrusters; selects the key performance characteristics of an electric propulsion system utilized in the Earth orbital, Lunar base, and Mars base analyses; and indicates a potential design for an electric propelled OTV.

E.1 DEVELOPMENT STATUS REVIEW

Electric propulsion systems hardware development began in the late 1950s and has continued at a relatively modest level since that time. Interspersed with this development have been several major flight tests of relatively small systems (less than 1 kWe), including those shown in Tables E-1 and E-2 for the USA.

There are several major types of electric propulsion systems that can be broadly categorized into electrothermal, electrostatic, and electromagnetic classes.

Electrothermal thrusters simply heat a propellant fluid and then accelerate it through a nozzle, in a process very similar to that of a nuclear rocket. Electrical heating of the propellant can be done using solid metal heaters or gaseous electrical arcs; thus, the attainable propellant temperatures are similar to those attainable in solid and/or gas-core nuclear rockets. Typical specific impulses attainable in electrothermal thrusters range no higher than 1500 sec and represent the low end of the range attainable with electric propulsion. Flight-qualified

TABLE E-1. SPACE FLIGHTS WITH ELECTRIC PROPULSION (1963-1975)

	USA	China, Japan, Europe	USSR
		ΔISAS, Electromagnetic	
1975	ΔATS 6, Ion		
			ΔMeteor 18, Electromagnetic Δ16 HAAZ, *Pulsed Plasma* Δ15 HAAZ, Pulsed Plasma
	ΔNavy Sat, Aug. N_2H_4 ΔNavy Sat. (4), Resistojet		ΔMeteor 10, Ion Electromagnetic
1970	ΔSERT 2, Ion		ΔYantar 4, Ion
	ΔATS 4, 5, Resistojet ΔLES 6, Pulsed Plasma		ΔYantar 3, Ion
		Legend: Δ Launched	ΔYantar 2, Ion
	ΔAdv Vela (4), Resistojet		
	ΔATS 1 & 3, Resistojet ΔNavy Sat. (5), Resistojet		ΔYantar 1, Ion
1965	ΔVela (2), Resistojet ΔSnapshot, Ion		
	ΔSERT 1, Ion ΔBlue Scout (3), Ion		ΔZond 2, Pulsed Plasma
1963			

TABLE E-2. SPACE FLIGHTS WITH ELECTRIC PROPULSION (1975-1985)

	USA	China, Japan, Europe	USSR
1985			Legend: Δ Launched
		ΔSpacelab 1, MPD	
	ΔSATCOM H, Aug. N_2H_4 ΔNova 3, Pulsed Plasma ΔSATCOM G, Aug. N_2H_4 ΔNova 2, Pulsed Plasma		No Data Available
		ΔMDT-2A, Pulsed Plasma	
	ΔNova 1, Pulsed Plasma ΔIntelsat V, Aug. N_2H_4	ΔETS 4, Pulsed Plasma	ΔMR, Pulsed Plasma
		ΔMS-T4, MPD	
1980			ΔGeo Rocket, MPD
			ΔMeteor, Electromagnetic
			ΔKust, MPD
		ΔK-9M-58 Electromagnetic	ΔMR, Pulsed Plasma
	ΔTIP 3, Pulsed Plasma		ΔMeteor, Electromagnetic ΔKosmos 780, MPD ΔKosmos 728, MPD
	ΔTIP 2, Pulsed Plasma		
1975			

electrothermal thrusters (resistojets) are in regular use for low thrust (0.1 lb) station keeping applications.

Electrostatic thrusters utilize electric fields to accelerate propellant ions generated in a special ionization chamber immediately upstream of the accelerator grids. Specific impulses range from a low of 1500 sec to a high in excess of 10,000 sec. Electrostatic thruster systems tend to be complex because of the ancillary equipment needed to generate the ions and to provide electrons for neutralization following the acceleration stage. The most well-known electrostatic thruster is the 30-cm mercury ion thruster developed by NASA-LERC and Hughes. Present development efforts are concentrating on using alternate propellants (xenon or argon) instead of mercury.

Electromagnetic thrusters utilize a combination of electric and magnetic fields to accelerate globally neutral, ionized gas plasmas in a manner similar to linear induction motors or MHD generators run in reverse. These devices generate specific impulses that range from 1500 to 8000 sec and operate at efficiencies somewhat lower than typical electrostatic thrusters. Electromagnetic thruster systems tend to be simpler than electrostatic thruster systems; however, trade-offs between the two are often done on the basis of lower mass versus higher performance.

One of the most important characteristics of electric propulsion systems is expressed in the relation:

$$T/P = N/2g\ Isp$$

where T is thrust, P is input power, n is total system electric to thrust power conversion efficiency, g is 9.8 m/s^2, and Isp is specific impulse. Higher specific impulses typically imply reductions in total propellant mass requirements. The above equation shows that higher specific impulse means more power is required for a given thrust. In other words, the propellant mass savings available with higher specific impulses is offset by the increased power supply mass required. This trade-off between propellant mass and power supply mass typically yields a minimum total mass

at specific impulses in the range of 1500 to 8000 sec for virtually any mission within the solar system.

Electric propulsion systems and associated electric power sources can be characterized by their specific mass, defined as the total system mass divided by the power. For an electric propulsion system, the power used in the total input power and typical specific masses range from 1 (advanced arcjet) to 30 (state-of-the-art ion thruster) kg/kWe. For a power source, the power is the total output power, and typical specific masses for future multimegawatt systems range down to 5 kg/kWe.

E.2 APPLICATIONS CHARACTERISTICS

Using optimistic values of thruster efficiency, power source specific mass and thruster system specific mass, it is possible to analytically derive the maximum vehicle acceleration attainable with electric propulsion. In the limit, where the vehicle is much larger than its payload and the propellant load small, the ultimately attainable acceleration for any electric propulsion system is less than 10^{-3} g. It is clear that electric propulsion cannot be used in any application requiring high acceleration. In particular, it cannot be used for surface-to-orbit propulsion on any body within the solar system except very small asteroids. This is a major area where nuclear rockets outperform electric propulsion.

The low accelerations available with electric propulsion preclude the use of minimum-energy Hohmann trajectories for transfers from one orbit to another. Electric-propelled vehicles typically follow spiral trajectories that require significantly more total propulsive energy. The necessary energy is characterized by the mission ΔV. For a Hohmann transfer from low Earth to geosynchronous orbit, the ΔV is 4160 m/s. For a low-thrust spiral trajectory, the ΔV is 5850 m/s. This difference partially offsets the potential savings available with higher specific impulse electric propulsion systems. In this example, the ΔV has increased by 41%, thus the electric propulsion specific impulse must be 41% higher than the comparable impulsive system, or no net propellant mass reduction will

result. If the comparable impulsive system is a nuclear rocket 900 sec specific impulse, a minimum specific impulse of 1,270 sec would be required for electric propulsion. This is easily met by more advanced electromagnetic or electrostatic thrusters, but could severely challenge nearer-term electrothermal systems.

The lower acceleration levels and higher energy requirements typical of electric propulsion systems and trajectories lead directly to system operating times (i.e., transfer times) that are much larger than those typical of impulsive systems. For example, a LEO-GEO transfer can require up to 100 days even with very high power systems. Longer trip times represent a significant limitation of electric propulsion system applications. In particular, any manned mission in cislunar space would not be done using electric propulsion because of the long-duration life support requirements implicit in multi-week flights. As another example, electric propelled transfer to GEO from LEO includes significant times spent in the Earth's Van Allen belts; thus, any payload sensitive to radiation would require more shielding than if an impulsive transfer were used.

Typically, interplanetary missions using impulsive rockets include spacecraft coast periods measured in months to years. For these long duration flights, electric propulsion systems can reduce flight times by thrusting continuously throughout what would otherwise be a coast period. A large manned Mars vehicle can transfer from the earth in times comparable to similar Hohmann trajectories. To the other planets, electric propulsion can reduce trip times by as much as 50%.

There are a number of technical issues associated with longer-duration missions typical of cislunar electric propulsion. For example, the very low propellant flow rates used result in storage requirements measured in months rather than hours. Long-duration storage of cryogens like argon is not yet entirely proven and would need to be, prior to the use of electric propulsion orbit transfer vehicles. Another example is that the power requirements for electric propulsion necessitate continuous source operation for long times. If the source is nuclear, the resultant

potential radiation exposure could be significant without additional shielding. Furthermore, the same long exposure concerns arise with propellant plume backflow contamination. While the propellant flow rates in electric propulsion systems are extremely small, the long exposure time can, if not carefully prepared for, lead to undesirable build-up of propellant contaminants on spacecraft surfaces. A related concern with the propellant plume is its ionization level and resulting interactions with microwave radar and communications systems.

One operational benefit associated with low acceleration, long duration trajectories is that a thruster failure is graceful, in that any uncontrolled thrust vectoring due to a failure could be shut down well before any significant trajectory excursions develop. Furthermore, typical prime propulsion system designs using electric propulsion utilize several complete thruster systems operating simultaneously. This virtually eliminates any concern with single point failures and allows the mission to continue, albeit in a degraded mode.

Electric orbit transfer vehicles under study today are intended for operating at power levels of megawatts. This is several orders of magnitude larger than existing system configurations. Development of higher power electric propulsion systems and power sources will require significant dollars and time. Based on development plans currently defined, a 100 ke system (SP-100 plus an ammonia arcjet) could be flight ready for demonstration in the mid-1990s. Megawatt-level systems will probably not be available until post-2005, at best. Development cost for these systems is difficult to estimate. Since a megawatt-level vehicle will probably consist of several thruster systems each running at several hundred kilowatts, the development of multimegawatt systems can be evolved from smaller systems. A 200 kWe MPD thruster system propulsion module for orbit transfer applications was estimated to require $55M to develop, excluding power. Production was estimated to cost about $20M. For comparison, NASA has developed the 2 kWe, 30-cm mercury ion engine system for a total investment since the early seventies of roughly $20M.

One interesting facet of high power electric propulsion system development arises with the expected test requirements. At megawatt power levels, the propellant flow rates are typically in the range of 5-10 g/s. Electric thrusters require environmental pressures of less that 10^{-5} Torr in order to operate properly. A ground test facility capable of maintaining 10^{-5} Torr pressure with an input flow of 5 g/s of argon would need to be roughly ten times as large as the present largest vacuum facility in the USA. Development of these high power thrusters may well need to be done in space in order to avoid overwhelming test facility costs.

Ultimately, comparison of electric propulsion to nuclear rocket systems will include parameters that are essentially not quantifiable. Issues are reliability, maintainability, versatility, overall development risk, safety and total system complexity to fully understand the future roles of these systems in space. Given the development challenges with either system, the ability to have a staged development with small evolutionary steps may prove to be a driving factor. In any event, it must be recognized that the present technical and programmatic understanding of either system is so limited that specific, detailed comparisons are doomed to be outdated as soon as they are completed.

E.3 ELECTRIC PROPULSION SYSTEM AND BASELINE VEHICLE SELECTION

Electric Propulsion (EP) systems are sized differently than chemical or nuclear systems because of low thrust considerations. Although there are a number of different electric propulsion technologies, previous studies have indicated that only ion and MPD thrusters are likely candidates for OTV applications. See Table E-3. Both technologies require a nuclear power system and have approximately the same Isp range.

There are differences in thruster efficiency and thruster specific mass. Figure E-1 shows a range of thruster efficiencies from other studies along with the efficiency selected for this study.

The thruster specific mass is the mass of the thruster assembly and power conditioning system per unit input power. Table E-4 shows the range

TABLE E-3. NEP SYSTEM OPTION PRELIMINARY SCREENING

Option	Applicability to Mission	Comments
Ion Thruster	Yes	--
Self-Field MPD Thruster	Yes	--
Applied-Field MPD Thruster	No	Too little data available. Performance similar to self-field MPD thruster.
Resistojets	No	Too low powered. Too low Isp.
Arcjets	No	Too low powered. Too low Isp. At higher power becomes a self-field MPD thruster.
Pulsed Inductive Thruster (PIT)	No	Pulsed power processing is life limited and heavy PIT performance does not justify added complexity.
Pulsed Plasma Thruster	No	Same as PIT.
Railguns	No	Same as PIT.
E-M Accelerators	No	Same as PIT.

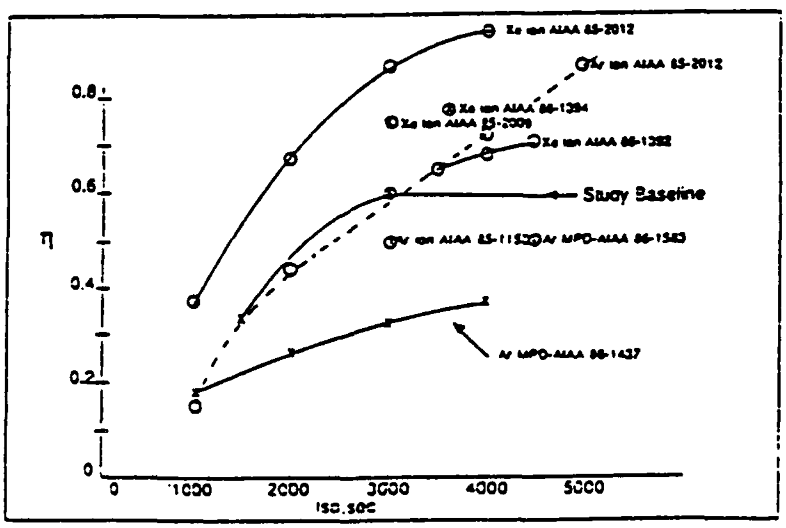

Figure E-1. NEP efficiency vs. specific impulse.

TABLE E-4. NEP SPECIFIC MASS

Thruster System	Specific Mass and Reference
Ar Ion	10.0 kg/kW - AIAA 86-1391 8.7 kg/kW - AIAA 82-1871
Xe Ion	10.0 kg/kW - AIAA 86-1391 8.4 kg/kW - AIAA 82-1871
Ar MPD	0.84 kg/kW - AIAA 86-1583 5.0 kg/kW - AIAA 86-1437 7.0 kg/kW - AIAA 82-1871
Power Source	
Advanced Nuclear	10.0 kg/kW

of specific masses from other reports. A thruster specific mass of 5 kg/kWe was selected for this study based on these values. A power system specific weight of 10 kg/kWe was selected to represent future power systems.

A characteristic of EP systems is the variation in thrust level with thruster Isp. The greater the Isp, the less the thrust. A high thrust EP OTV will use more propellant; however, the transfer time will decrease. This is represented in Figures E-2 and E-3 for the transfer of a 14,000-lb payload to GEO, with Isp of 3000 sec and 4000 sec, respectively.

The 3000-sec and 4000-sec Isp systems have nearly the same dry weight because the power system and thruster weight make up the majority of the vehicle weight. There are small differences in propellant tank weight. The 3000-sec Isp uses more propellant, but the transfer time is less. Figures E-2 and E-3 display another characteristic of EP systems in that increased power decreases transfer time only up to a point. The reason is that although the increase in power does increase thrust, the power system weight also increases. A point is reached where the payload becomes a small part of total weight and the vehicle consists only of thrusters and power supply.

If deployment of the payload were not time critical, an EP OTV would tend to have very low power with very high Isp to save propellant weight. However, most payloads need to be deployed in a reasonable length of time. If that time is specified, then the user only needs to go to a chart such as Figure E-2 or E-3 and select the power level that meets the required deployment time.

A second reason for decreasing deployment time is the size of the required OTV fleet. Since a low power of high Isp EP OTV will require a longer time to deploy a payload, it will deploy fewer payloads over its life. Although it will use less propellant to do so, the increased cost of additional OTVs may very well drive up the overall cost. Figure E-4 shows the relationship between OTV power level and Isp to the size of the OTV fleet.

An accepted method of determining optimum power level takes the time cost of the payload value into account. The payload must be designed and fabricated some time prior to its required deployment. If the deployment transfer time is long, the payload must be fabricated even earlier. With inflation at a specified annual rate, the earlier the payload is fabricated, the more it costs. The cost associated with the deployment transfer time can be found by taking the product of the actual payload value and the inflation rate (referred to as the discount rate). For a $140M payload, and a discount rate of 10% per year, the deployment time costs $38.3K per day. This cost must be added to the launch mass cost to get an accurate comparison between heavy, short-duration-transfer chemical/nuclear stages and light, long-duration-transfer EP-OTVs.

One method of adding the transfer time cost to the launch mass cost is to convert transfer time days into equivalent mass and adding this to the known launch mass. This can be done by dividing the transfer time cost per day by the launch cost/lb and results in a conversion factor in lb/day. Table E-5 illustrates the process and shows the resulting value of 51 lb/day. The net effective launch mass then becomes the actual launch mass plus the equivalent transfer time mass, as shown.

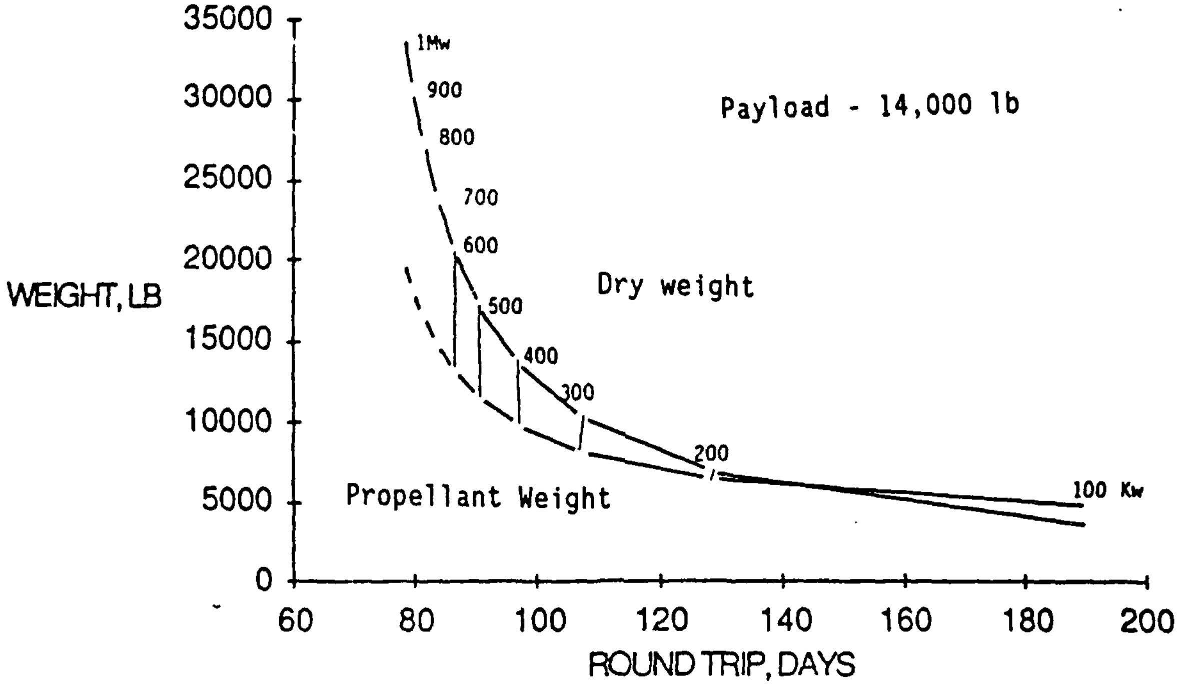

Figure E-2. NEP LEO-GEO propellant and dry weight vs. transfer time (Isp = 3000).

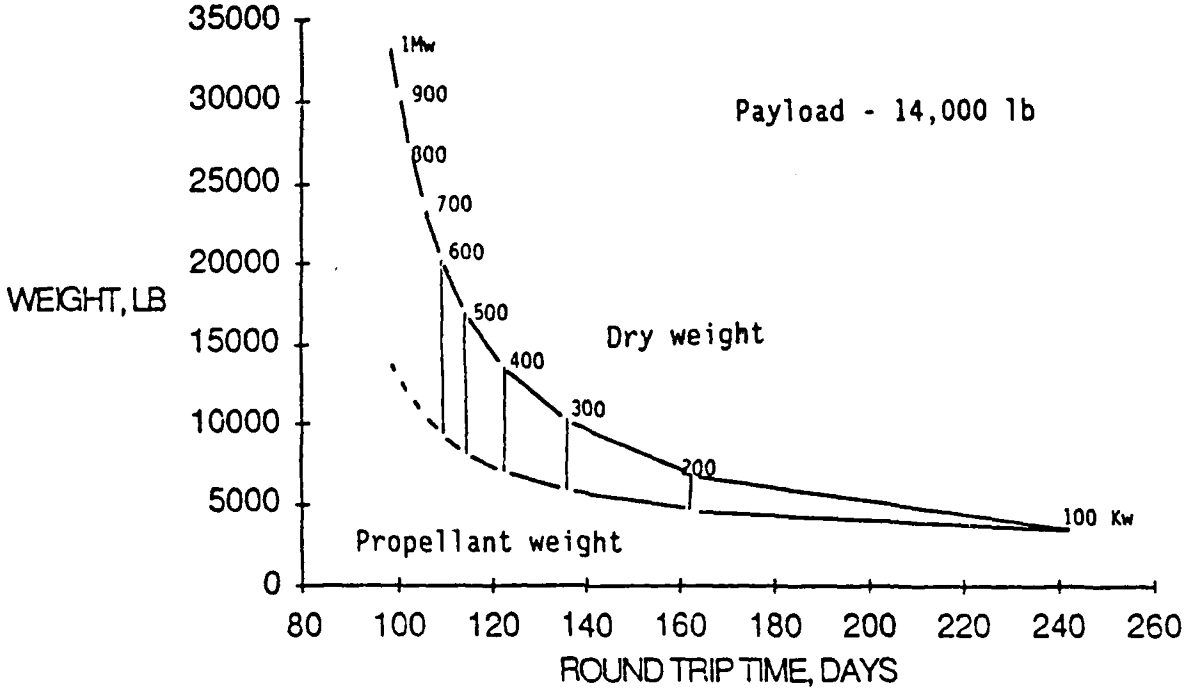

Figure E-3. NEP LEO-GEO propellant and dry weight vs. transfer time (Isp = 4000).

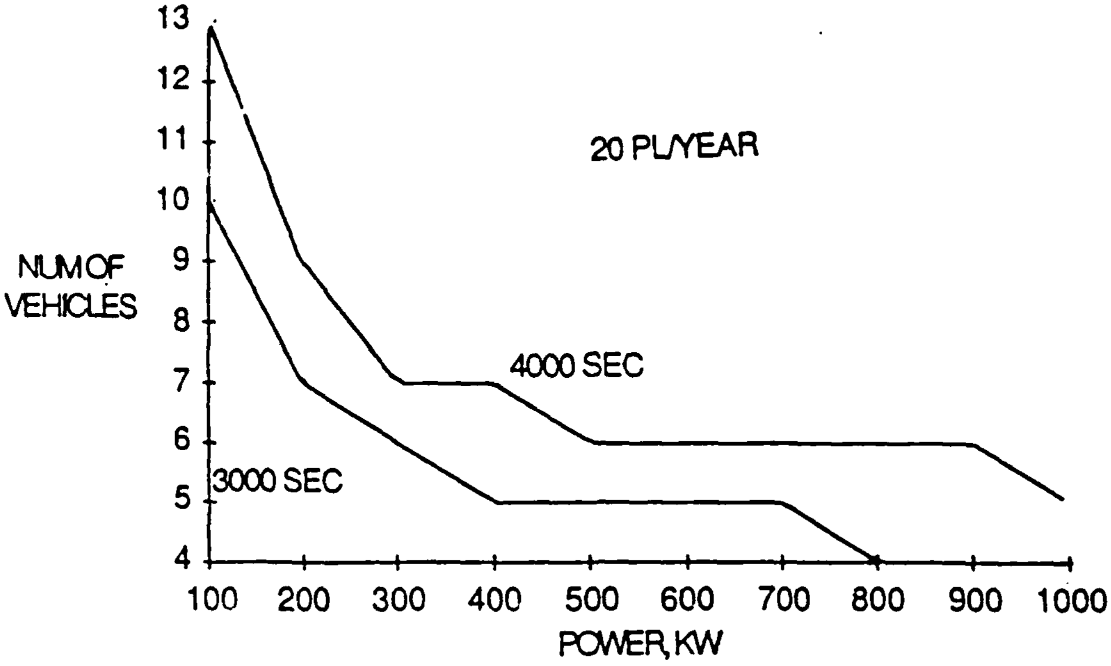

Figure E-4. NEP OTV fleet size vs. power level.

TABLE E-5. NEP DEPLOYMENT TIME SELECTION

Method to Determine Power Level that Optimizes Launch Cost + Transfer Cost

- Transfer Cost = Spacecraft Cost x Interest Rate x Deploy Time
- Transfer Cost = \$140M x 10% per Year x Deploy Time = 38.8K \$Day
- Transfer Cost can be Converted to Equivalent lb of Launch Mass

$$\frac{\text{Transfer Cost}}{\text{Launch Cost}} = \frac{38.3\text{K \$/Day}}{750 \text{ \$/lb}} = 51 \text{ lb/Day}$$

- Effective Launch Mass (lb) = Actual Launch Mass (lb) + 51 (lb/Day) x Deploy Time (Days)

Once a deployment cost factor is determined, it is a simple matter to determine the power level that minimizes cost in terms of effective launch mass. Figure E-5 shows how effective launch mass is a function of power system output and Isp for a reusable EP-OTV fleet delivering 5 or 20 payloads per year to GEO. This figure shows that as power level decreases, life cycle effective launch mass decreases until the transfer cost begins to drive the effective launch mass higher. The power level that gives the minimum effective launch mass is optimum. This figure includes the dry weight of the fleet but not the replacement of thruster assemblies. The figure indicates that a power level of 300 kWe at 4000 Isp is optimum. A power level of 500 kWe was selected as the baseline for this study. This is somewhat greater than Figure E-5 indicates as optimum; however, the curve is nearly flat in this region and the inclusion of the cost of the fleet would tend to drive the "bucket" to higher power level (less vehicles required at higher power levels). In addition, this baseline OTV is also used for Lunar and Mars missions which also tend to favor higher power levels.

To summarize, a baseline EP-OTV was selected that has a power level of 500 kWe. A specific power of 10 kg/kWe and a thruster specific mass of 5 kg/kWe result in a base dry weight of 16,500 lb. A propellant tank

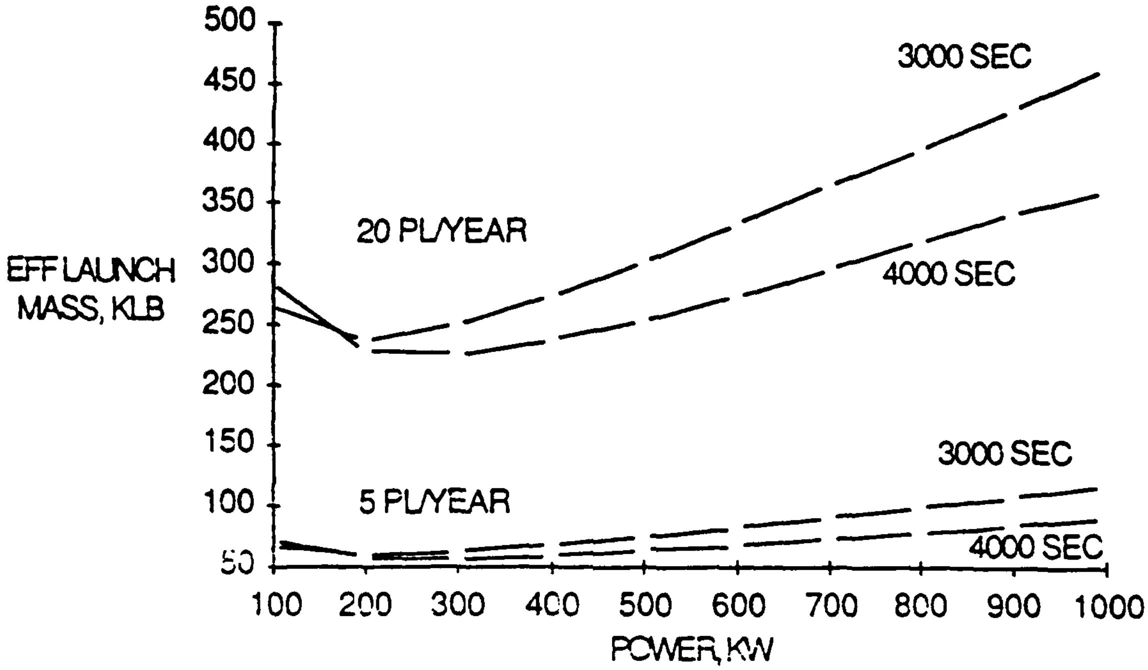

Figure E-5. NEP OTV optimum power level.

weight and unusable propellant weight of 6% of propellant weight were selected. The Isp is 4000 with a thruster efficiency of 60%. Figure E-6 shows how this EP-OTV design could look.

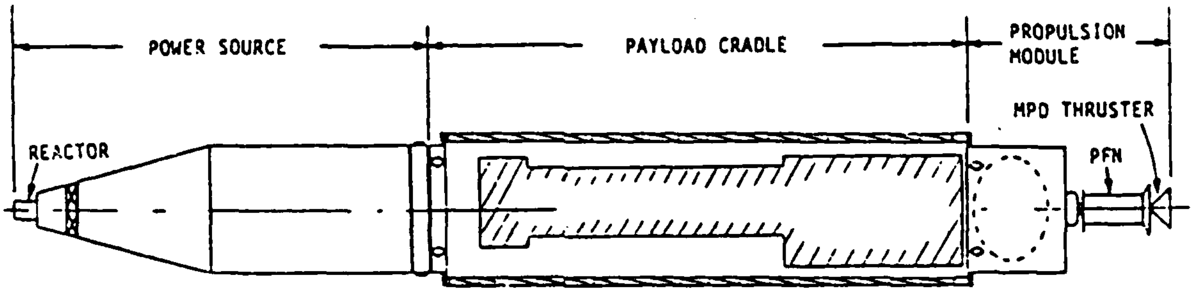

Figure E-6. Nuclear-electric OTV design.

APPENDIX F

PARAMETRIC ANALYSIS GRAPHICAL RESULTS

APPENDIX F
PARAMETRIC ANALYSIS GRAPHICAL RESULTS

This appendix provides the graphical results for the LEO-GEO nuclear stage parametric analysis discussed in Section 4.4. The figures included herein are as follows:

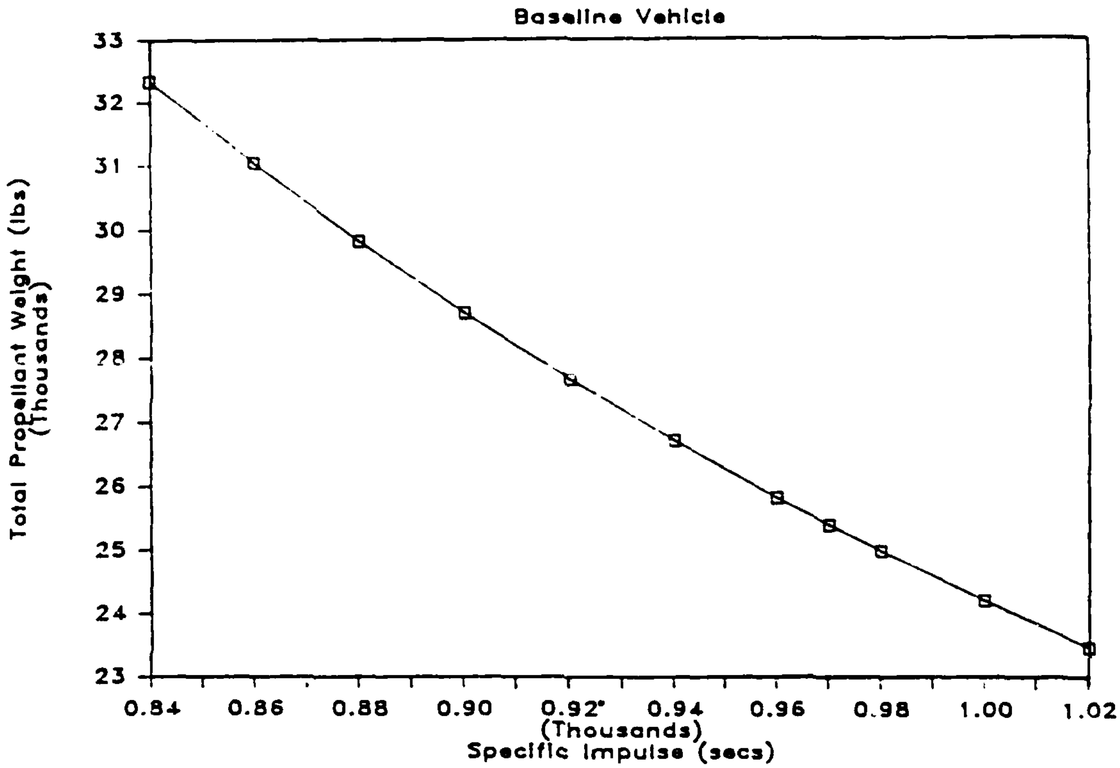

Figure F-1. Total propellant weight vs. specific impulse.

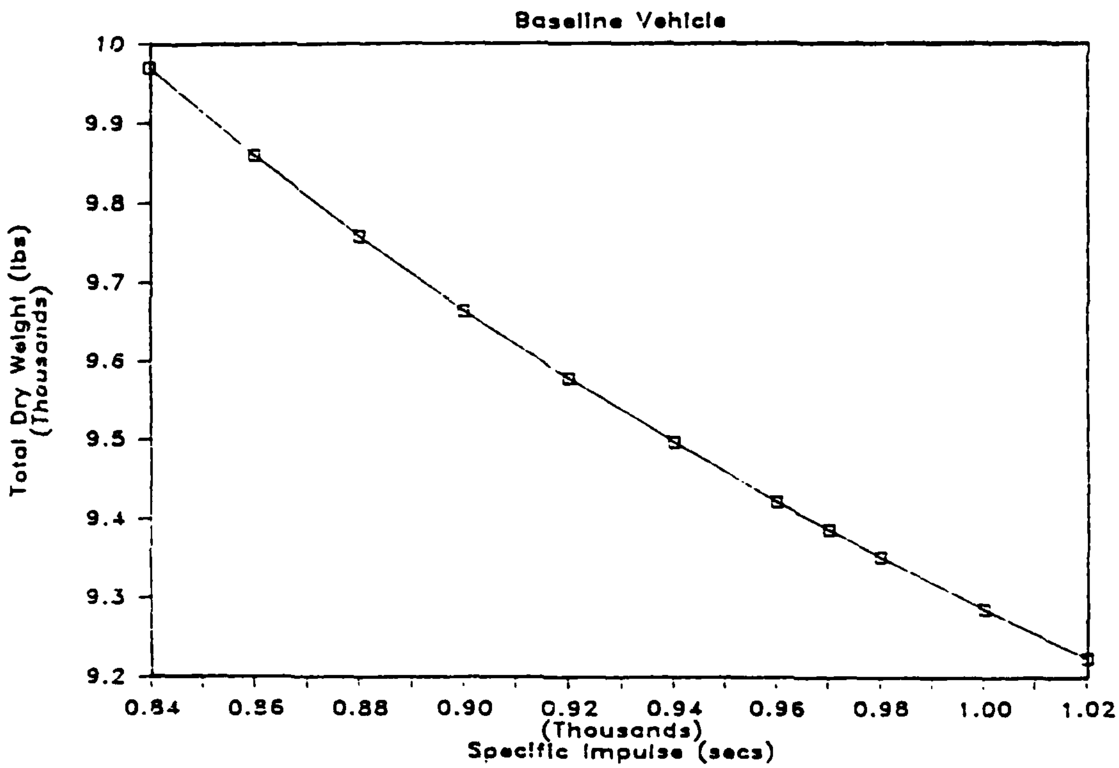

Figure F-2. Dry weight vs. specific impulse.

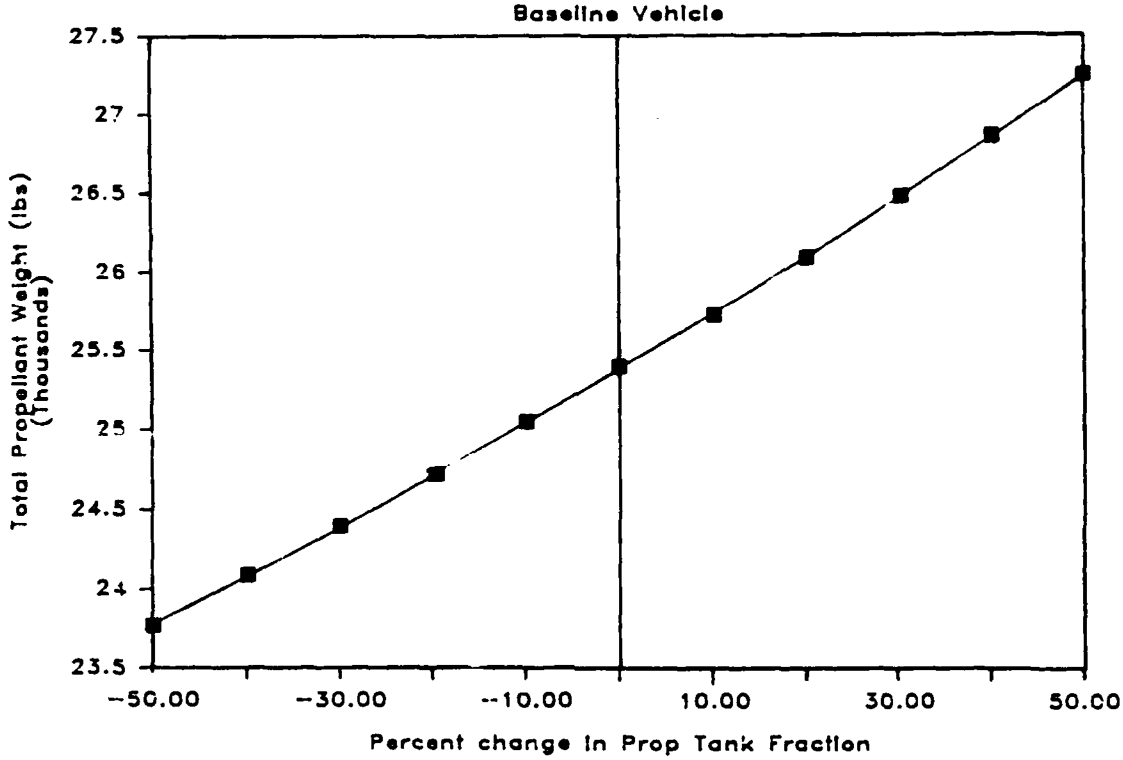

Figure F-3. Total propellant weight vs. propellant tank fraction.

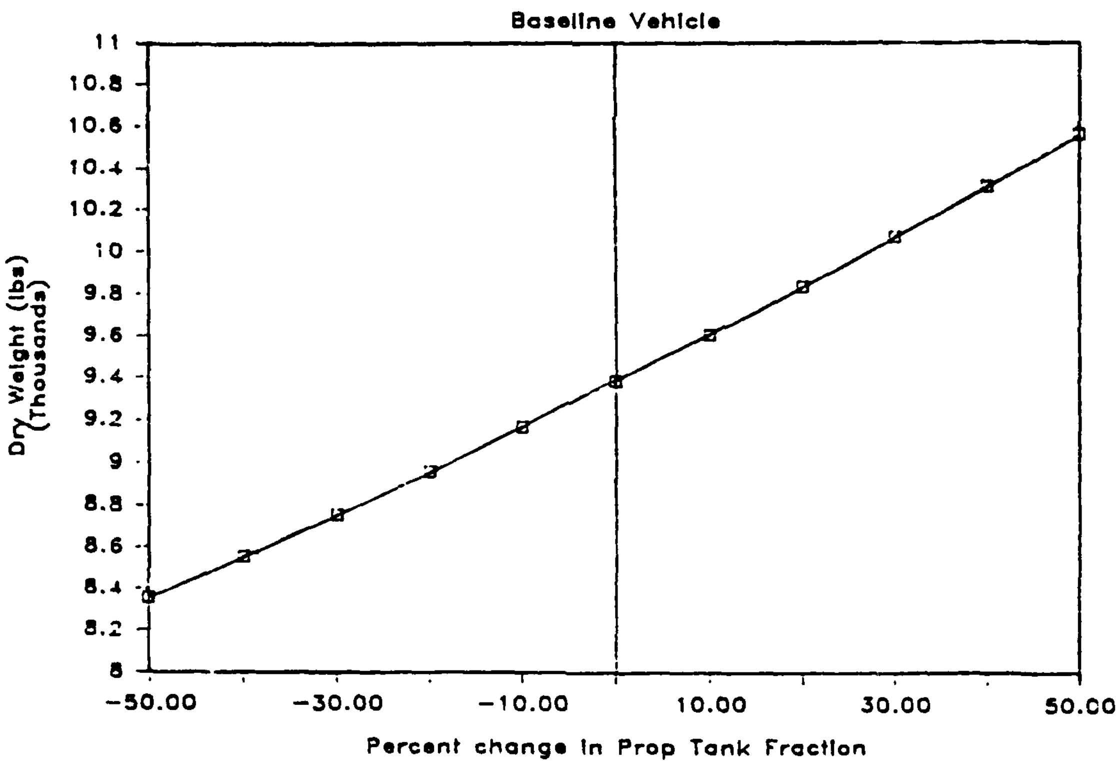

Figure F-4. Dry weight vs. propellant tank fraction.

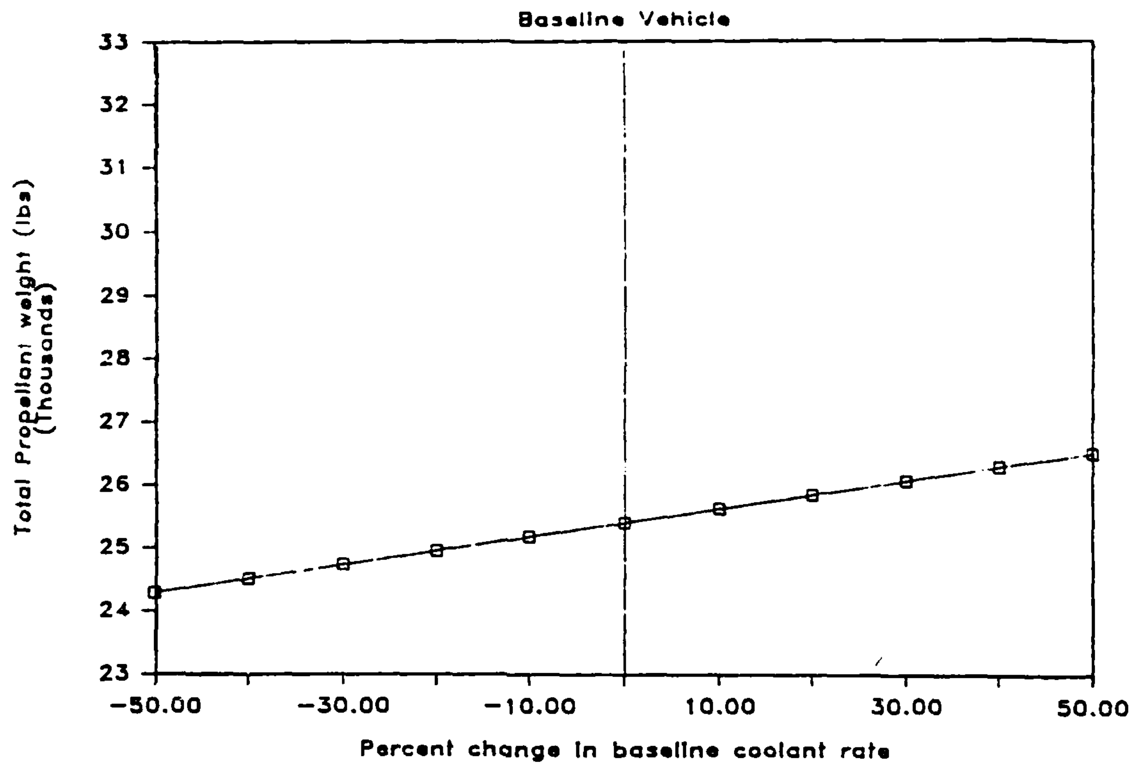

Figure F-5. Total propellant weight vs. cooling rate.

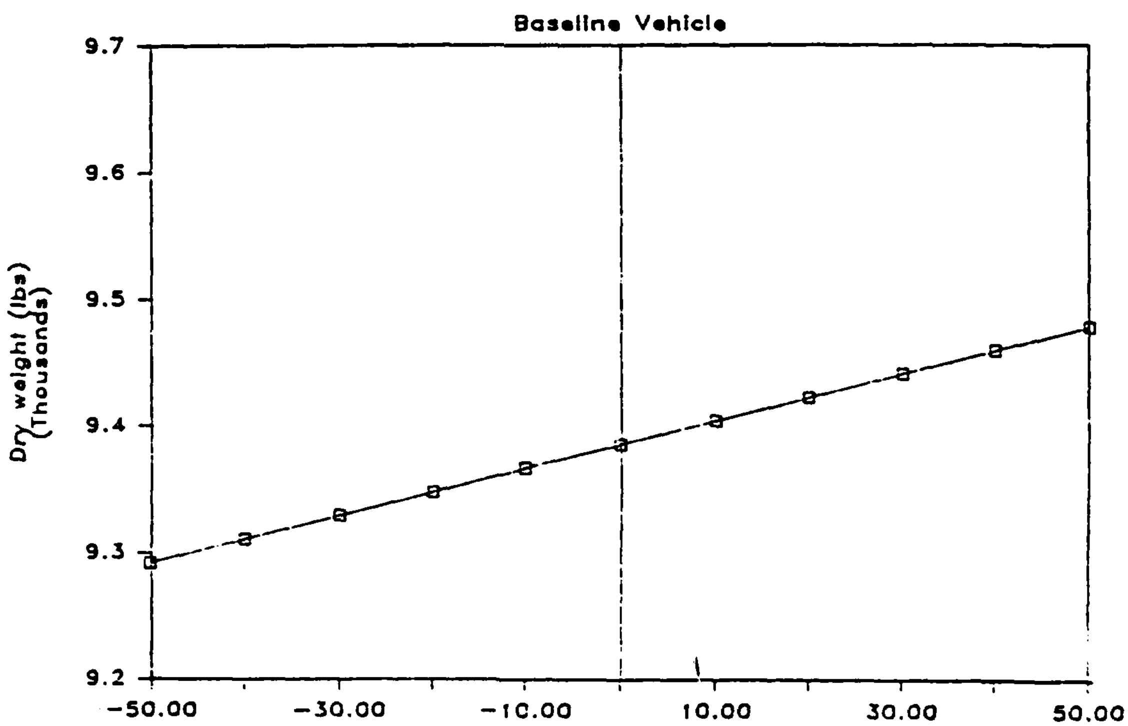

Figure F-6. Dry weight vs. cooling rate.

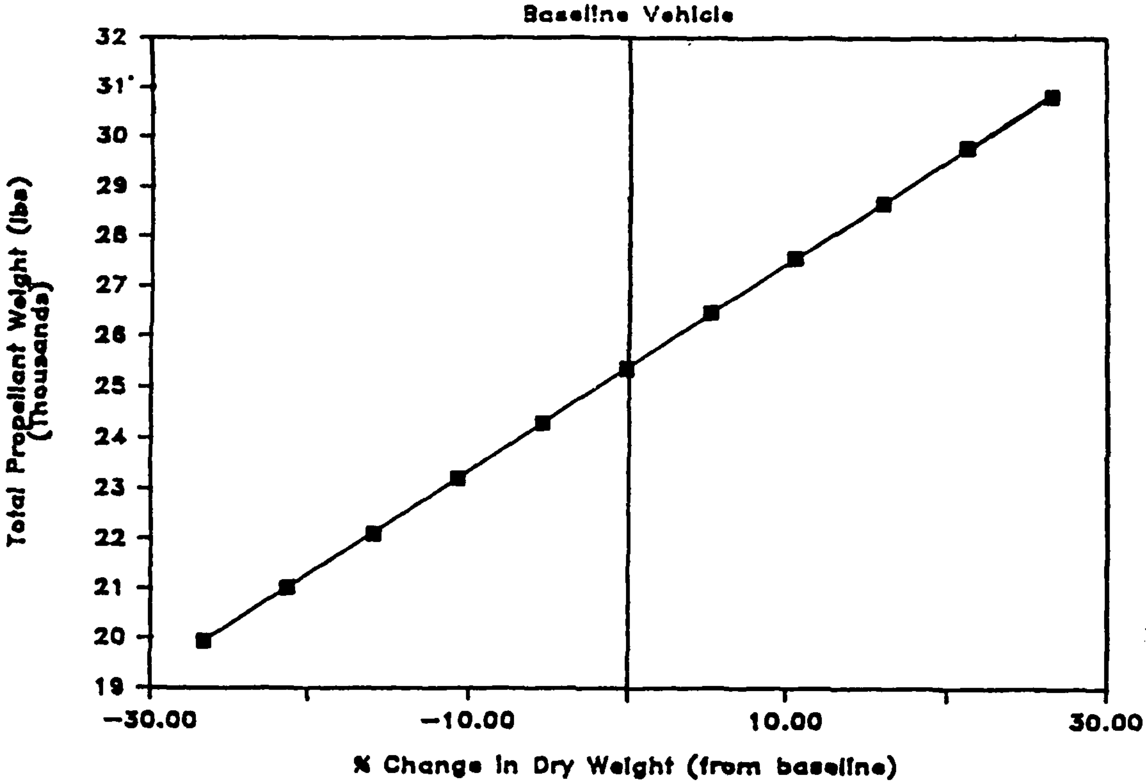

Figure F-7. Total propellant weight vs. dry weight change.

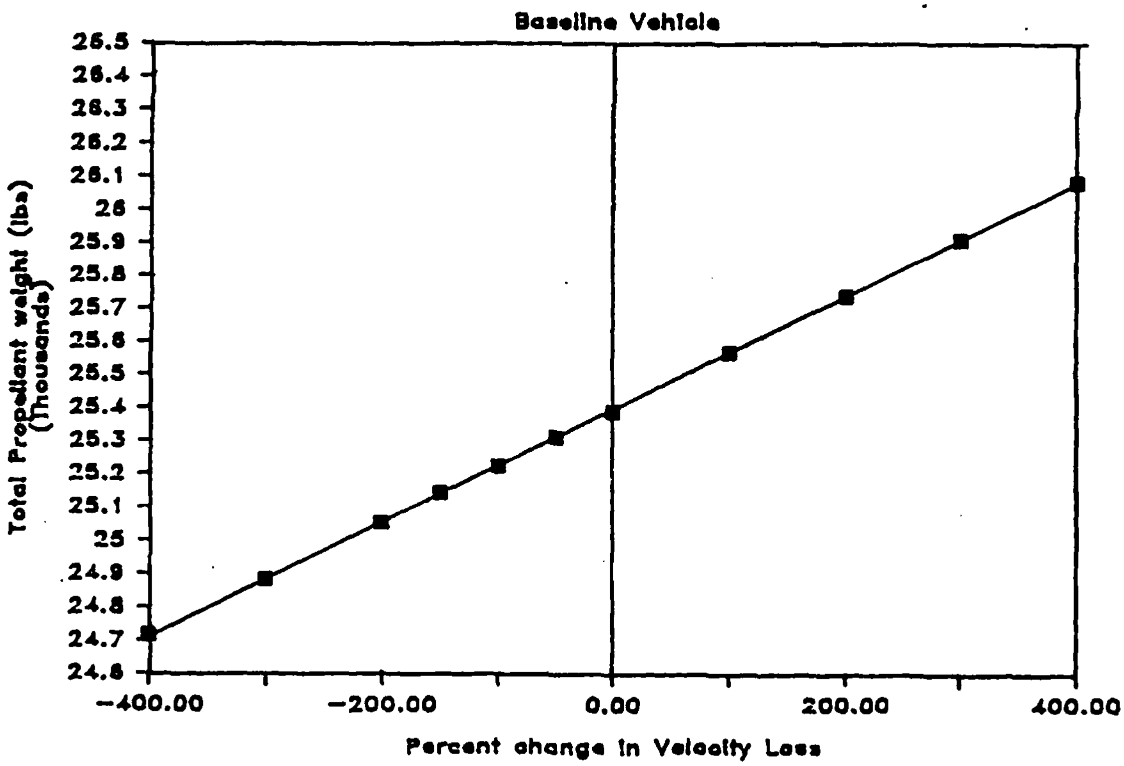

Figure F-8. Total propellant weight vs. velocity loss.

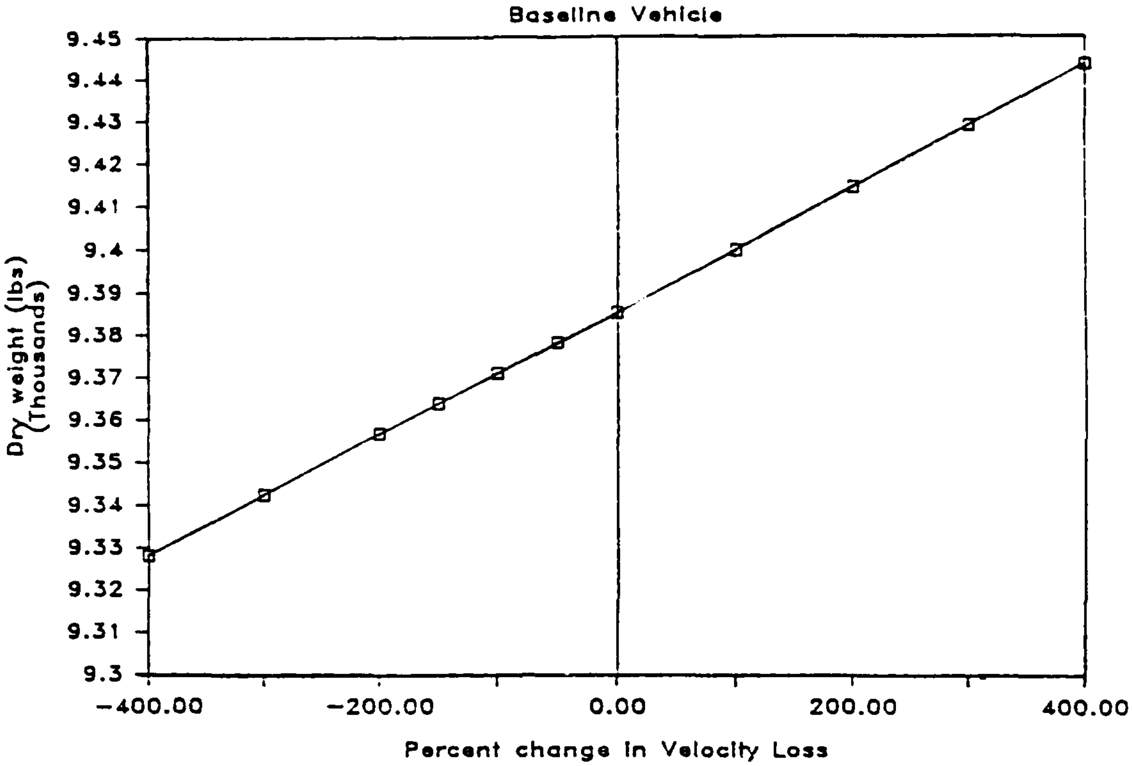

Figure F-9. Dry weight vs. velocity loss.

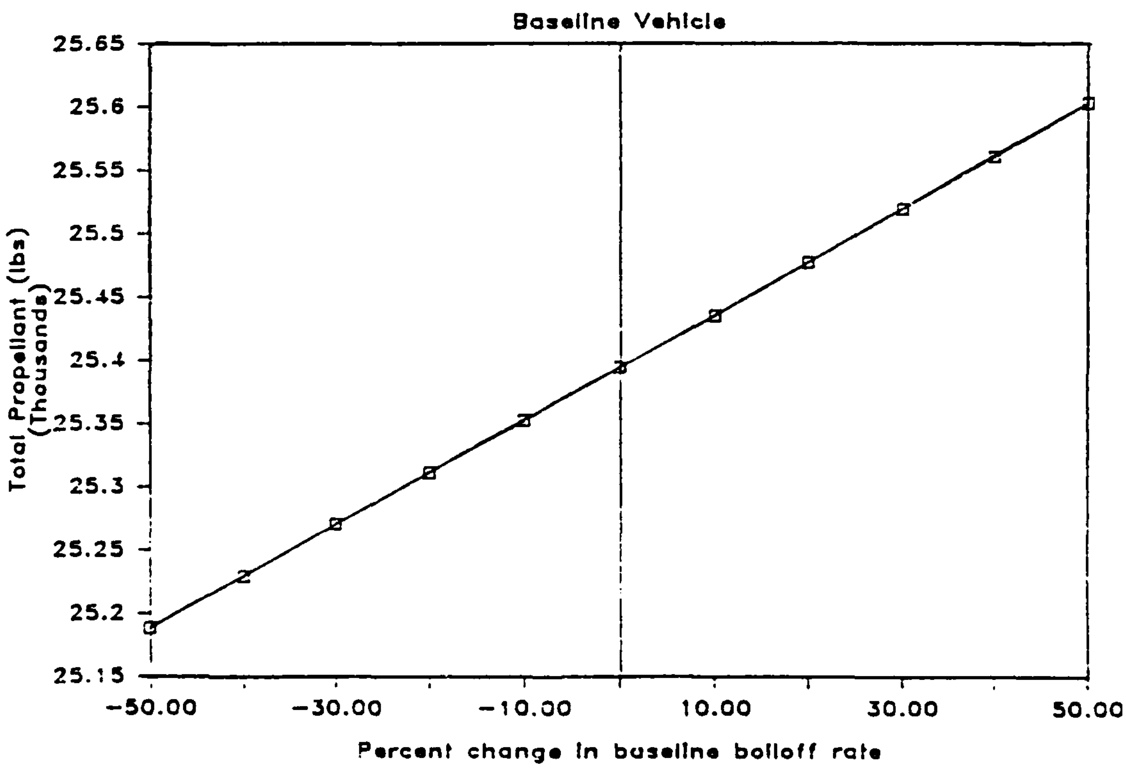

Figure F-10. Total propellant weight vs. boiloff rate.

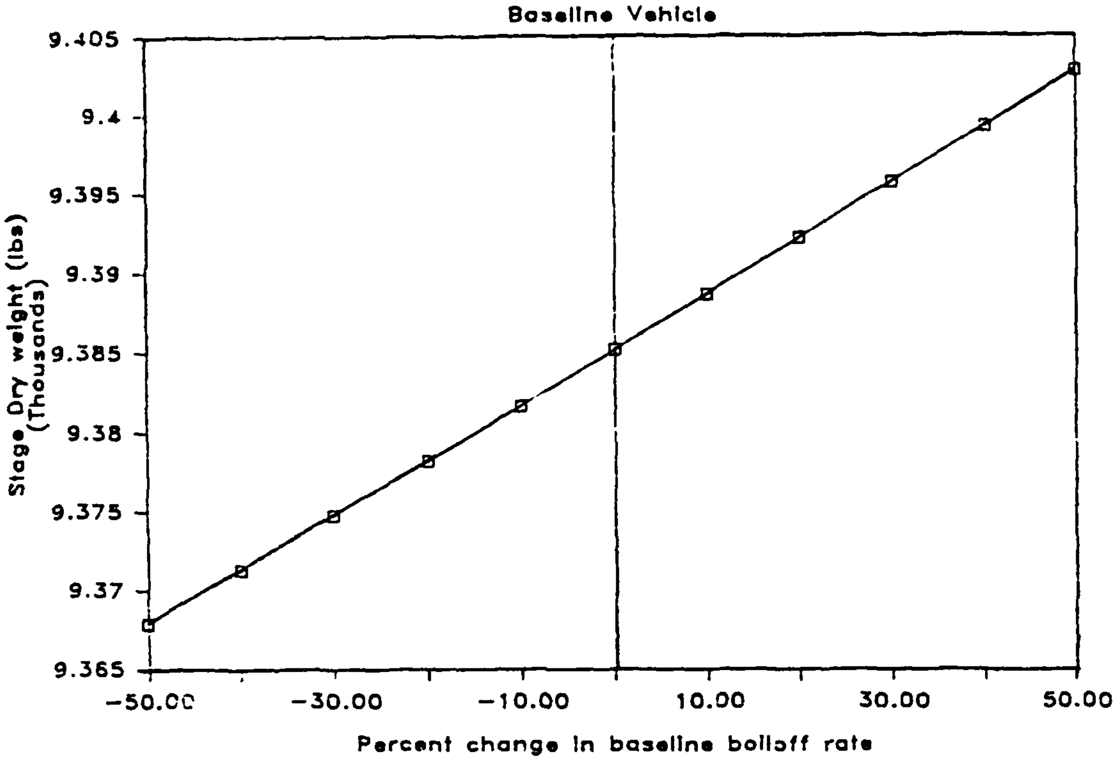

Figure F-11. Dry weight vs. boiloff rate.

APPENDIX G

GEO STAGE COST BREAKDOWN

APPENDIX G
GEO STAGE COST BREAKDOWN

This appendix provides the stage cost breakdowns that were used in the operational cost sensitivity analysis (Section 4.5) and the life-cycle-cost (LCC) analysis (Section 4.6). The estimated basic costs are provided in the following tables:

Table G-1. GEO Stage Design, Development Test, and Evaluation Costs

Table G-2. GEO Stage Production Costs

Table G-3. GEO ANRE Stage Operations Costs

Table G-4. GEO Chemical Stage Operations Costs

Table G-5. GEO Nuclear Electric Propulsion Stage Operations Costs

TABLE G-1. GEO STAGE DESIGN, DEVELOPMENT, TEST AND EVALUATION COSTS, $M

Design and Development

	Nuclear	Chemical	Electrical
Structures	22.8	24.8	26.8
Tanks	15	17	14
Propulsion	12.6	12.6	12.6
Main Engine	850	275	900
RCS	11.6	11.6	11.6
GN&C	81.5	81.5	92.3
C&DH	39.4	39.4	42.6
EPS	16.6	16.6	16.6
GSE	5.2	5.2	5.2
ASE	10.3	15.3	12.3
SSE	0	0	0
	1065	499	1134
Software	73	63	68
Tooling	27	27	27
System Test	349.5	176.7	368.7
Systems Engineering	272.61	137.826	287.586
Program Management	178.711	90.352	188.528
	1965.821	993.878	2073.814

TABLE G-2. GEO STAGE PRODUCTION COSTS (2 UNITS), $M

Flight Hardware

	Nuclear	Cryogenic	Electrical
Structures	3.78	3.78	5.58
Propellant Tanks	4.86	5.58	4.50
MPS (without engines)	5.04	5.04	4.50
Main Engines (with reactor)	51.84	10.80	72.00
ACS	3.96	3.96	3.96
GN&C	10.80	10.80	18.00
C&DH	21.60	21.60	27.00
Electrical Power	3.78	3.78	3.42
Thermal/Meteor Shield	2.52	2.88	2.16
A & C/O	14.40	9.00	9.00
	122.58	77.22	150.12
Tooling	12.26	7.72	15.01
Sustaining Engineering	14.71	9.27	10.01
Systems Engr & Integ	3.68	2.32	4.50
Program Management	8.58	5.41	10.51
	161.81	101.93	198.16

TABLE G-3. GEO ANRE STAGE OPERATIONS COSTS

Space Based Nuclear Stage Operations (millions of 1986 dollars)
Based on the old III/3 Mission Model

Number of Missions:	20	Engine Life:	80
$/lb to Orbit:	750	Structure Life:	100
Engine Cost ($M):	28.8	Struc Cost ($M):	39
Propellant Wt (lb):	25394	P/L Capacity (lb):	14000
Stage Reliability:	0.995	% Prop Scavenged:	60%
Dry Weight (lb):	9385	Engine Wt (lb):	4600
# of Deliveries:	1	Structure Wt (lb):	4785
Stage OPS Factor:	1.00	SB Accomm Factor:	1.00
Learning Curve:	0.90	B:	-0.15

Operations	CPF	Total Ops	First Unit Cost
Stage Operations			
Aerobrake	0.00	0.00	0.00
Engines	1.01	20.30	1.60
Propulsion	0.08	1.52	0.12
RCS	0.08	1.52	0.12
Structures	0.06	1.27	0.10
Tanks	0.10	1.90	0.15
GN&C	0.13	2.54	0.20
G&DH	0.25	5.07	0.40
Power	0.05	1.01	0.08
Thermal	0.04	0.76	0.06
ASE	0.03	0.51	0.04
GSE	0.02	0.38	0.03
SSE	0.01	0.25	0.02
	1.85	37.04	2.92
Refurb Costs			
IVA	0.32	6.34	0.50
EVA	0.00	0.00	0.00
GB Manpower	0.10	1.90	0.15
Misc Spares	0.75	15.00	0.00
Mission Ops	0.16	3.17	0.25
	1.32	26.42	0.90

TABLE G-3. (continued)

	CPF	Total Ops	First Unit Cost
Launch Costs			
Delivery	0.35	7.04	0.33
Eng Spares	0.04	0.86	0.04
Brk Spares	0.00	0.00	0.00
Struc Spares	0.04	0.72	0.04
Return	0.00	0.00	0.00
	0.43	8.62	0.41
Propellant Costs			
GB Acquisition	0.20	4.00	0.20
SB Del Tanker	7.62	152.36	7.62
"FREE" Prop	3.05	60.95	3.05
	10.87	217.31	10.87
SB Accom Costs			
Tank Farm	0.85	17.00	1.34
Hangar	0.00	3.04	0.24
OMV Use	0.01	15.35	1.21
RFB Hardware	0.00	2.79	0.22
Software	0.20	2.54	0.20
Utilities	0.00	0.00	0.00
Misc IVA/EVA	0.00	0.00	0.00
	1.06	40.72	3.21
P/L Costs			
P/L Attach	0.05	1.00	0.05
P/L Processing	0.10	2.00	0.10
P/L User Chg	0.25	5.00	0.25
P/L Transport	10.50	210.00	10.50
	10.90	218.00	10.90
Program Support	2.26	45.30	2.68
Mission Loss	0.15	2.97	0.15

TABLE G-3. (continued)

	CPF	Total Ops	First Unit Cost
Totals			
Stage Ops	1.85	37.04	2.92
Refurb	1.32	26.42	0.90
Launch	0.43	8.62	0.41
Propellant	10.87	217.31	10.87
Accom	1.06	40.72	3.21
P/L	10.90	218.00	10.90
Program Support	2.26	45.30	2.68
Mission Loss	0.15	2.97	0.15
	28.84	596.36	32.04

TABLE G-4. GEO CHEMICAL STAGE OPERATIONS COSTS

Space Based Cryogenic Stage Operations (millions of 1986 dollars)
Based On The Old III/3 Mission Model

Number of Missions:	20	Engine Life:	10
$/lb to Orbit:	750	Structure Life:	40
Engine Cost ($M):	6	Struc Cost ($M):	35
Propellant Wt (lb):	53000	P/L Capacity (lb):	14000
Stage Reliability:	0.995	% Prop Scavenged:	60%
Dry Weight (lb):	10132	Engine Wt (lb):	792
# of Deliveries:	1	Structure Wt (lb):	9340
Stage OPS Factor:	0.75	SB Accomm Factor:	0.90
Learning Curve:	0.90	B:	-0.15

Operations	CPF	Total Ops	First Unit Cost
Stage Operations			
Aerobrake	0.00	0.00	0.00
Engines	1.01	20.30	1.60
Propulsion	0.06	1.14	0.09
RCS	0.06	1.14	0.09
Structures	0.05	0.95	0.08
Tanks	0.07	1.43	0.11
GN&C	0.10	1.90	0.15
G&DH	0.19	3.81	0.30
Power	0.04	0.76	0.06
Thermal	0.03	0.57	0.05
ASE	0.02	0.38	0.03
GSE	0.01	0.29	0.02
SSE	0.01	0.19	0.02
	1.64	32.85	2.59
Refurb Costs			
IVA	0.32	6.34	0.50
EVA	0.00	0.00	0.00
GB Manpower	0.10	1.90	0.15
Mics Spares	1.48	29.50	0.00
Mission Ops	0.16	3.17	0.25
	2.05	40.92	0.90

TABLE G-4. (continued)

	CPF	Total Ops	First Unit Cost
Launch Costs			
Delivery	0.38	7.60	0.33
Eng Spares	0.06	1.19	0.04
Brk Spares	0.00	0.00	0.00
Struc Spares	0.18	3.50	0.04
Return	0.00	0.00	0.00
	0.61	12.29	0.41
Propellant Costs			
GB Acquisition	0.20	4.00	0.20
SB Del Tanker	15.90	318.00	15.90
"FREE" Prop	6.36	127.20	6.36
	22.46	449.20	22.46
SB Accom Costs			
Tank Farm	0.85	17.00	1.34
Hangar	0.00	3.04	0.24
OMV Use	0.01	15.35	1.21
RFB Hardware	0.00	2.79	0.22
Software	0.18	2.54	0.20
Utilities	0.00	0.00	0.00
Misc IVA/EVA	0.00	0.00	0.00
	1.04	40.72	3.21
P/L Costs			
P/L Attach	0.05	1.00	0.05
P/L Processing	0.10	2.00	0.10
P/L User Chg	0.25	5.00	0.25
P/L Transport	10.50	210.00	10.50
	10.90	218.00	10.90
Program Support	4.08	81.56	4.39
Mission Loss	0.22	4.38	0.22
Totals			
Stage Ops	1.64	32.85	2.59
Refurb	2.05	40.92	0.90

TABLE G-4. (continued)

	CPF	Total Ops	First Unit Cost
Launch	0.61	12.29	0.41
Propellant	22.46	449.20	22.46
Accom	1.04	40.72	3.21
P/L	10.90	218.00	10.90
Program Support	4.08	81.56	4.37
Mission Loss	0.22	4.38	0.22
	42.00	879.92	45.06

TABLE G-5. GEO NUCLEAR-ELECTRIC-PROPULSION STAGE OPERATIONS COSTS

Space Based Cryogenic Stage Operations (millions of 1986 dollars)
Based on the old III/3 Mission Model

Number of Missions:	20	Thrust Life:	5
$/lb to Orbit:	750	Power System & Structure Life:	40
Engine Cost ($M):	40	Struc Cost ($M):	50
Propellant Wt (lb):	7347	P/L Capacity (lb):	14000
Stage Reliability:	0.995	% Prop Scavenged:	60%
Dry Weight (lb):	16941	Thrust Wt (lb):	1500
		Power System:	15000
# of Deliveries:	1	Structure Wt (lb):	441
Stage OPS Factor:	0.80	SB Accomm Factor:	0.90
Learning Curve:	0.90	B:	-0.15

Operations	CPF	Total Ops	First Unit Cost
Stage Operations			
Aerobrake	0.00	0.00	0.00
Engines	1.01	20.30	1.60
Propulsion	0.06	1.22	0.10
RCS	0.06	1.22	0.10
Structures	0.05	1.01	0.08
Tanks	0.08	1.52	0.12
GN&C	0.10	2.03	0.16
G&DH	0.20	4.06	0.32
Power	0.04	0.81	0.06
Thermal	0.03	0.61	0.05
ASE	0.02	0.41	0.03
GSE	0.02	0.30	0.02
SSE	0.01	0.20	0.02
	1.68	33.69	2.66
Refurb Costs			
IVA	0.32	6.34	0.50
EVA	0.00	0.00	0.00
GB Manpower	0.10	1.90	0.15
Mics Spares	9.25	185.00	0.00
Mission Ops	0.16	3.17	0.25
	9.82	196.42	0.90

TABLE G-5. (continued)

	CPF	Total Ops	First Unit Cost
Launch Costs			
Delivery	0.64	12.71	0.33
Eng Spares	0.23	4.50	0.04
Brk Spares	0.00	0.00	0.00
Struc Spares	0.29	5.79	0.04
Return	0.00	0.00	0.00
	1.15	23.00	0.41
Propellant Costs			
GB Acquisition	0.20	4.00	0.20
SB Del Tanker	2.20	44.08	2.20
"FREE" Prop	0.88	17.63	0.88
	3.29	65.71	3.29
SB Accom Costs			
Tank Farm	0.85	17.00	1.34
Hangar	0.00	3.04	0.24
OMV Use	0.01	15.35	1.21
RFB Hardware	0.00	2.79	0.22
Software	0.18	2.54	0.20
Utilities	0.00	0.00	0.00
Misc IVA/EVA	0.00	0.00	0.00
	1.04	40.72	3.21
P/L Costs			
P/L Attach	0.05	1.00	0.05
P/L Processing	0.10	2.00	0.10
P/L User Chg	0.25	5.00	0.25
P/L Transport	10.50	210.00	10.50
	10.90	218.00	10.90
Program Support	2.37	47.49	1.51
Mission Loss	0.16	3.13	0.16
Totals			
Stage Ops	1.68	33.69	2.66
Refurb	9.82	196.42	0.90

TABLE G-5. (continued)

	CPF	Total Ops	First Unit Cost
Launch	1.15	23.00	0.41
Propellant	3.29	65.71	3.29
Accom	1.04	40.72	3.21
P/L	10.90	218.00	10.90
Program Support	2.37	47.49	1.51
Mission Loss	0.16	3.13	0.16
	30.41	628.15	23.03

APPENDIX H

ANRE - IMPACT OF REDUCED SPECIFIC IMPULSE

APPENDIX H
ANRE - IMPACT OF REDUCED SPECIFIC IMPULSE

The baseline ANRE configuration delineated in the main report assumed a specific impulse value of 970 sec. This value is based upon advanced technology for fuel elements and materials in the reactor. This appendix provides comparative data for additional configurations at other values of specific impulse. Because the original NERVA systems demonstrated 850 sec, a slightly greater value (870 sec) was used as the minimum for this analysis. An intermediate value of 900 sec was also analyzed. The resulting alternative configurations, based upon the LEO to GEO mission, are compared to the baseline configuration by weight and by cost.

WEIGHT ANALYSIS

The configuration sizing for the two alternate values of specific impulse proceeded in the manner described in Section 4. The selected mission was the LEO to GEO delivery. Only the delivered engine performance was altered while all other ground rules remained the same.

As would be expected, the configurations with the lower specific impulse values are both heavier in dry and total propellant weight. Tables H-1 and H-2 are weight statements for the two alternate configurations. Figures B-1 and B-2 of Appendix B demonstrate the change in total dry weight and total mission propellant for a range of specific impulse, including the above selected values.

COST ANALYSIS

A cost sensitivity analysis was performed to determine the impacts of varying the specific impulse values from 970 sec to 900 and 870 sec. Figure H-1 shows that as the specific impulse increases from 870 to 970 sec the 20-mission life cycle cost decreases from $2.8B to $2.75B. This trend is entirely due to the amount of propellant saved by going to a higher specific impulse. The 5 Klb of propellant saved by having a higher specific impulse generates a savings of slightly over $40M over the

20 missions for the baseline case. However, it was assumed that the DDT&E and unit production costs between the three cases would be essentially equal. If this ground rule were to change, the cost savings would probably be reduced because the higher specific impulse engine would be more costly to develop and produce.

It is interesting to note that, even for the lowest specific impulse case (870 sec) for the nuclear stage, over 2 Klb of propellant per mission are saved over the chemical stage, and a total of about $100M in life cycle cost (20 mission baseline) is saved over that of the electrical stage.

TABLE H-1. WEIGHT STATEMENT - NUCLEAR LEO-GEO STAGE - 870 SEC ISP
(All Weight in lb)

Gross Flight Weight	54,212	Total Propellant	30,405
Cooling Propellant	1,000	Boiloff	217
Dry Weight	9,808	Tankage Weight	2,280
	Stage Length (ft) 55.70		

TABLE H-2. WEIGHT STATEMENT - NUCLEAR LEO-GEO STAGE - 900 SEC ISP
(All Weight in lb)

Gross Flight Weight	52,364	Total Propellant	28,700
Cooling Propellant	1,000	Boiloff	207
Dry Weight	9,664	Tankage Weight	2,152
	Stage Length (ft) 53.27		

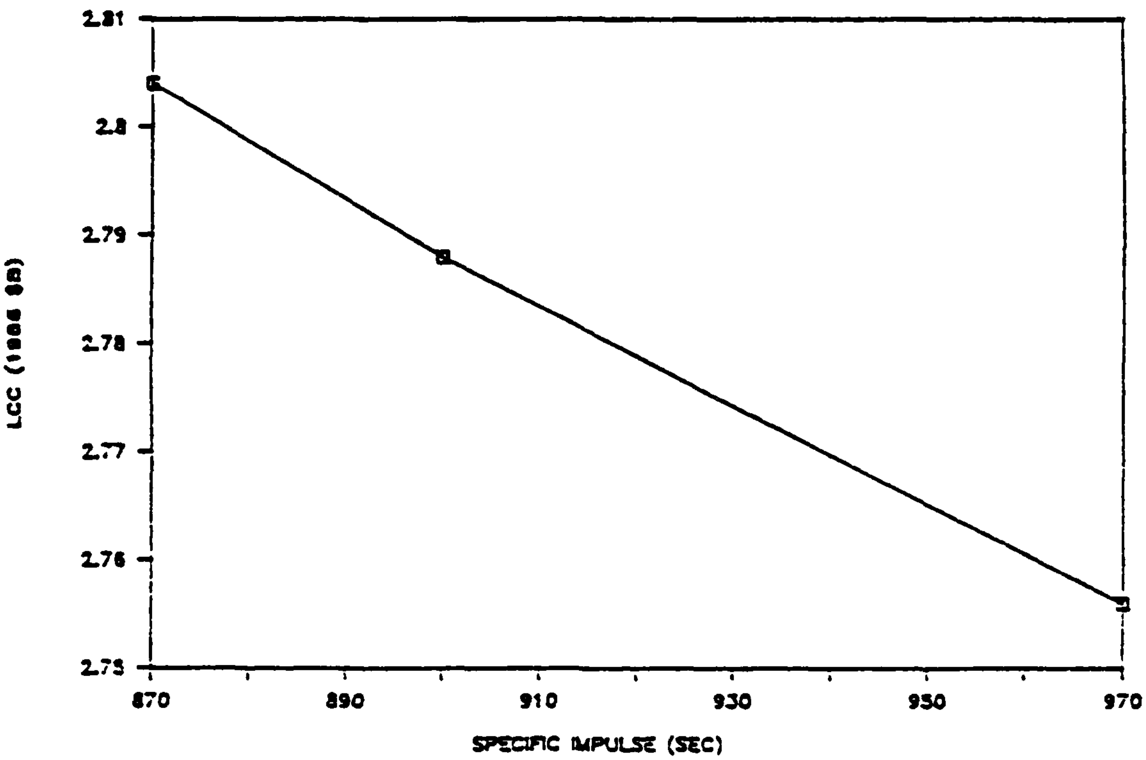

Figure H-1. 20-mission life cycle cost vs. specific impulse.

APPENDIX I

SAFETY OF THE ANRE IN EARTH ORBIT TRANSFER MISSIONS

APPENDIX I

SAFETY OF THE ANRE IN EARTH ORBIT TRANSFER MISSIONS

There are a number of safety and reliability issues associated with utilization of a nuclear rocket, or tug, for Earth orbit transfer missions. Most of these issues concern failures that can result in one of the following adverse situations:

1. Contamination of the biosphere

2. Contamination of the service platform

3. Contamination of the spacecraft

4. Loss of engine

5. Loss of mission.

The first two are safety related since they could be life-threatening (it is assumed that the service platform is manned and tug is unmanned); the last three are reliability issues. This appendix identifies the failures that can result in each of these situations and proposes measures that can be taken to mitigate the adverse consequences.

A transfer mission is a complex operation involving various elements in addition to the tug. In analyzing potential failure modes, therefore, it is necessary to address a complete transfer cycle and consider the actions of all of the participating elements. This is the approach taken herein, based on the description of a typical transfer mission presented in Section 4.1 of the main body of this report.

A summary is provided of those actions that must be successfully performed by each of the mission elements if the mission is to succeed. A review of flight safety issues includes a summary and analysis of results of a NERVA safety study conducted by Aerojet Nuclear Systems Company.[1]

Finally, failure modes, consequences, and mitigating procedures are identified for each of the adverse situations.

I.1 ROLE OF MAJOR MISSION ELEMENTS

The various elements that are involved in a transfer mission are described, along with the function performed, below.

II.1 Central Control

It is assumed that control of the entire operation is centralized. The control station could be located on the service platform, at a separate location in space, or on the Earth. This element has responsibilities in the following areas.

II.1.1 Surveillance

1. Identify potential problem areas in the total space environment. The control station must have data concerning all of those objects whose actions could impact operations; this includes not only the other orbital transfer elements but any object that could influence the transfer.

2. Locate tug, robot, spacecraft, rendezvous point (intercept point for the tug and spacecraft, for example), service stations and ultimate destination.

3. Monitor the status of performance sensors located on the tug, robot, spacecraft, and service platform.

4. Monitor the status of backup devices.

5. Perform necessary computations concerning required actions (response to performance sensors located on tug, robot, spacecraft, and service platform).

II.1.2 Communication

1. Maintain communications with orbital traffic control center. To ensure success of the operation, a data base--referred to as the traffic control center--must exist that contains knowledge of the total space environment.

2. Maintain contact with tug, robot, spacecraft, and service platform.

II.1.3 Control

1. Respond to potential problem areas in the total space environment

2. Control disengagement from service platform

3. Guide to rendezvous point

4. Control docking

5. Guide to ultimate destination

6. Control disengagement from spacecraft

7. Guide to service platform

8. Control attachment to service platform.

II.1.4 Command

Develop programs to direct the operations.

II.2 Orbital Traffic Control Center

This element has real-time data concerning all elements that could impact the success of the orbital transfer. It also has the capability to

duplicate the functions listed above that are performed by central control. This control center will probably be located on the Earth, although it could also be on the platform on another location in space. To minimize the possibility of being rendered nonoperational along with central control, these two stations should not be co-located.

II.3 Spacecraft (Payload)

This is the item to be moved. It is assumed to be unmanned and could be a satellite or a variety of other objects. Its responsibilities can be summarized as follows:

1. Maintain acceptable docking mechanism.

2. Maintain acceptable configuration. The object must maintain its integrity and refrain from actions which could interfere with the mission (such as unplanned maneuvers, deployment of appendages during docking, or transmissions which could interfere with control).

3. Respond to guidance from the control station, which requires that the communication system remains operational and that internal control mechanisms are functional.

4. Maintain acceptable performance monitors.

II.4 Service Platform

This is an element in a circular orbit around the Earth that is used for maintenance and as a transfer station. It has the same requirements as the spacecraft. In addition, it must perform the following functions:

1. Provide refueling capability
2. Install spacecraft
3. Provide service and maintenance
4. Perform checkout prior to launch.

I1.5 Tug (OTV)

This is the element that provides the power for Earth orbit transfers. For this analysis, the tug is broken into six subsystems, most of which are patterned after the "Small Engine" nuclear rocket design illustrated in Figure I-1. Although the details will probably change, it is assumed that the Advanced Nuclear Rocket Engine will have components similar to the following:

I1.5.1 Propellant Feed Subsystem

This element begins with the liquid hydrogen tank shown at the top of Figure I-1. It includes a single-stage centrifugal pump and a single-stage turbine. It contains the following five valves and actuators:

1. Propellant shutoff valve (PSOV)--located at the bottom of the propellant tank and provides a tight seal against propellant leakage when the engine is not in use.

2. Nozzle control valve (NCV)--used to adjust the flow split between the nozzle coolant tubes and the tie-tubes.

3. Turbine discharge block valve (TDBV)--isolates the turbine during preconditioning and cooldown.

4. Turbine bypass control valve (TBCV)--regulates the amount of flow to the turbine and, consequently, the turbopump speed and flow rate.

I1.5.2 Nozzle Assembly, Pressure Vessel, and Closure Assembly

The nozzle assembly consists of the nozzle and nozzle extension. The nozzle contains coolant tubes and is cooled by propellant during operation. The pressure vessel houses and provides support for the nuclear subsystem and contains the high pressure hydrogen propellant fluid which flows through the nozzle and around the enclosed reflector to the internal

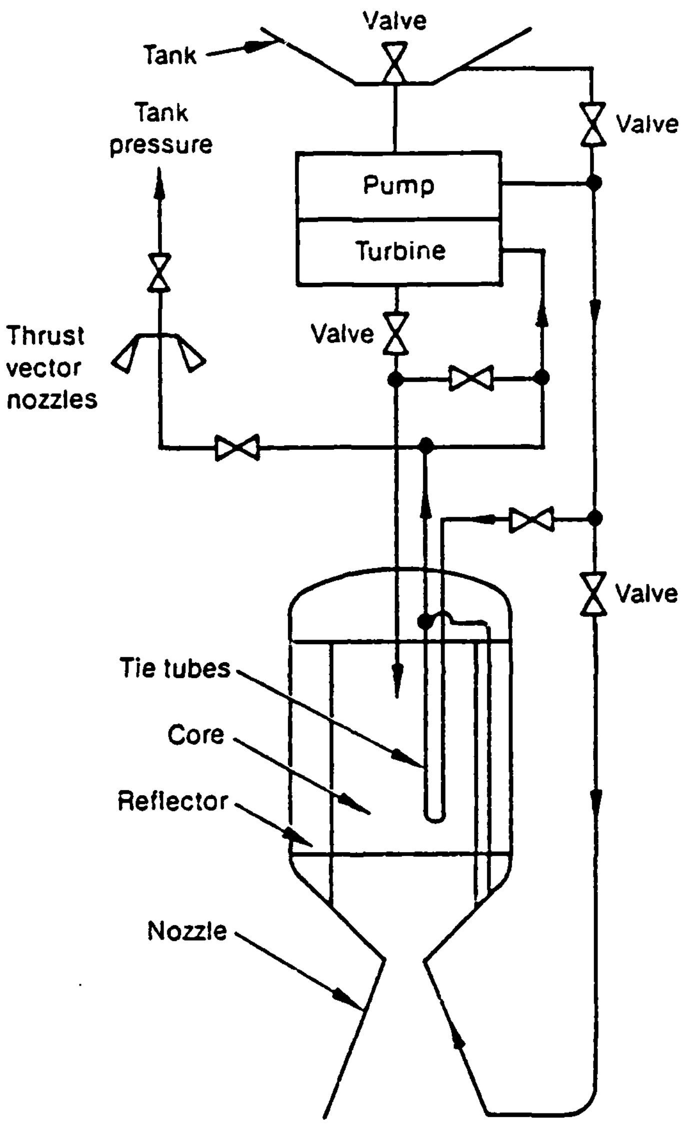

Figure I-1. Advanced nuclear rocket engine schematic.

shield. The assembly also transmits thrust from the nozzle to the lower thrust structure.

I1.5.3 Thrust Structure and External Shield

The thrust structure is divided into an upper thrust structure (UTS) and a lower thrust structure (LTS). The UTS transmits loads from the gimbaled portion of the engine to the vehicle interface and provides support for fluid lines and the electrical harness. The LTS transmits engine static and dynamic loads during boost and thrust operations from the pressure vessel closure to the gimbal assembly. It also provides support for components of the propellant feed subsystem, structural support coolant lines, the wiring harness, and the disk shield.

I1.5.4 Gimbal Assembly

The gimbal assembly consists of the gimbal structure that interfaces with the upper and lower thrust structures and the gimbal actuators. The gimbal actuators are servo-control systems consisting of an electromechanical actuation mechanism and a control amplifier. The gimbal actuators provide thrust vector position control about the pitch and yaw axes.

I1.5.5 Reactor Assembly

The reactor assembly consists of a nuclear reactor and an actuation system for reactivity control devices with associated instrumentation and controls. The reactor consists of fuel elements, a core periphery, support plates and plena, an internal shield, a reflector assembly, and control drum drive assemblies. Reflector coolant is provided from the nozzle coolant channel exhausts. The support stem coolant exhaust is mixed with the reflector coolant flow at the reflector outlet and is used as drive power for the engine turbopumps. The turbine exhaust gas flows through the dome flow baffle, internal shield, plenas between the core support plate and the internal shield and reactor core, and through the reactor core.

This gas is heated by the reactor assembly to operating temperatures and exhausted out the nozzle.

II.5.6 Performance Sensors, Communication, and Control Equipment

II.6 Robot (Optional)

This is a small maneuvering element; it is normally attached to the tug and has the following missions:

1. Retrieve spacecraft and attach to tug at the service platform

2. Move tug to and from service platform

3. Detach tug from spacecraft and move to safe distance after orbital transfer

4. Perform selected maintenance on tug

5. Monitor performance of tug and spacecraft.

I.2 FLIGHT SAFETY AND RELIABILITY ISSUES

The information in Tables I-1 through I-4 below was extracted from NERVA safety studies conducted by Aerojet Nuclear Systems Company;[2] the system design that they evaluated is illustrated in Figure I-2. The following failure definitions are from the same report:

1. Category III: Failures that result in inability of the engine to meet its normal-mode performance of service-life requirements but which allow Emergency Mode Operation or single turbopump operation. Failures in this category are further subdivided as follows:

    a. Category IIIA: Failures that require Single Turbopump Operation

TABLE I-1. SINGLE FAILURE POINT SUMMARY - FAILURE EFFECT CATEGORY IIIA NONNUCLEAR SUBSYSTEMS

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
Propellant Shutoff valve (PSOV)	Closed	2	1018.0
	Intermediate	2	24.0
Turbine Block Valve (TBV)	Closed	2	1018.0
	Intermediate	2	22.0
Turbine Throttle Valve (TTV)	Closed	2	976.0
Turbine Discharge Block Valve (TDBV)	Open	2	94.0
	Closed	2	970.0
	Intermediate	2	22.0
Pump Discharge Check Valve (PDKV)	Closed	2	882.0
	Intermediate	2	10.0
Turbopumps (TPA)	Will Not Start	2	680.0
	Stops	2	100.0
	Low Performance	2	1180.0
Turbine Inlet Line (TBV to TPA)	External Leakage	2	14.1
Turbine Discharge Line (TPA to TDBV and TTV)	External Leakage	2	14.2
Pump Inlet Line (PSOV to TPA)	External Leakage	2	16.1
Pump Discharge Line (TPA to PDKV)	External Leakage	2	14.1
	Total	34	7054.5

TABLE I-2. SINGLE FAILURE POINT SUMMARY - FAILURE EFFECT CATEGORY IIIB

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
Nonnuclear Subsystems			
SPKV	Internal Leakage	2	4.0
Nuclear Subsystems			
One Cluster SPT Plate Attachment Hardware	Mechanical Failure, No Fragments	1	TBD[a]
One SPT Stem Liner	Mechanical Failure - No Fragments	1	TBD
	Mechanical Failure - Fragments	1	TBD
Support Plate	Rupture/Fracture of Bypass Plenum/ Core Inlet Plenum Interface	1	TBD
	Rupture/Fracture of Bypass Plenum/ Mixing Plenum Interface	1	TBD
	Rupture/Fracture of Mixing Plenum/ SPT Plate Axial Flow Passage Interface	1	TBD
	Rupture/Fracture of Bypass Flow Passages/Reflector Outlet Plenum Interface	1	TBD

TABLE I-2. (continued)

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
Bypass Piping Distribution Torus	Ruptures/Fractures	1	TBD
Flow Baffle (SPT Plate and Plena)	Excessive Joint Leakage	1	TBD
Flow Diffuser (SPT Plate and Plena)	Rupture/Fracture	1	TBD
	Yields/Bows	1	TBD
One Bypass Flow Pipe	Ruptures/Fractures	1	TBD
	Shears/Decouples	1	TBD
One Stem Feed Pipe	Ruptures/Fractures	1	TBD
	Shears/Decouples	1	TBD
Central Shield	Lead Layer Thickness Decreases	1	TBD
	Bath Mechanical Failure	1	TBD
	Bath Spalling	1	TBD
Peripheral Shield	Lead Layer Thickness Decreases	1	TBD
	Bath Mechanical Failure	1	TBD
	Bath Spalling	1	TBD

TABLE I-2. (continued)

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
One Control Drum	Fails in Place	1	TBD
	Decouples From CDOA-Bias Spring Available	1	TBD
	Decouples From CDOA-Bias Spring Unavailable	1	TBD
	Loss of Poison Plate	1	TBD
One AMOD	Fails to Maintain Mechanical coupling of Actuator and Control Drum-AMOD Unavailable	1	TBD
TOTAL NONNUCLEAR SUBSYSTEMS		2	4.0
TOTAL NUCLEAR SUBSYSTEMS		26	TBD
TOTAL NUCLEAR AND NONNUCLEAR		28	TBD

a. To be determined

TABLE I-3. SINGLE FAILURE POINT SUMMARY FAILURE EFFECT CATEGORY IV NONNUCLEAR SUBSYSTEMS

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
SPKV	Open	2	68.0
	Intermediate	2	8.0
CSL (Coolant line MPT to CSOVs)	External leakage	1	13.1
CL (Coolant line CSCVs to SSCL)	External leakage	1	5.1
SPSL (TDL to MRT)	External leakage	1	2.2
PDL and SSBL (PDKV to nozzle and pressure vessel)	External leakage	1	20.1
TIL (Pressure vessel to TBVs)	External leakage	1	13.1
TDL (TDBV and TTV to pressure VES)	External leakage	1	13.2
TBL (BCVs to TDL)	External leakage	1	5.1
Upper Thrust Structure	Does not transmit loads	1	22.0
Lower Thrust Structure	Does not transmit loads	1	13.0
External Shield	Mechanical failure	1	16.0
Pressure Vessel and Closure	Catastrophic structural failure and gross leakage	1	24.0
Nozzle Extension	Excessive erosion, flange failure, etc.	1	403.0

TABLE I-3. (continued)

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
Nozzle (Main)	Tube leakage, blockage, etc.	1	467.0
Gimbal Assembly	Actuator motor failure, mechanical failure, etc.	1	242.0
	Total	18	1345.9

TABLE I-4. SINGLE FAILURE POINT SUMMARY-FAILURE EFFECT CATEGORY IV NUCLEAR SUBSYSTEMS

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
SSCL (PDL to pressure vessel)	External leakage	1	12
One Fuel Element	Mechanical failure	1	TBD
One Central Element	Mechanical failure fragments created	1	TBD
One Support Stem	Mechanical failure fragments created	1	TBD
One CHES	Mechanical failure fragments created	1	TBD
	Mechanical failure fragments ejected	1	TBD
Filler Strips	Fracture--Aft end	1	TBD
Support Plate	Rupture/fracture of feed plenum/SPT plate inlet plenum interface	1	TBD
	Rupture/fracture of feed plenum/ mixing plenum interface	1	TBD
	Rupture/fracture of feed plenum/ reflector outlet plenum interface	1	TBD
	Yields/bows	1	TBD
	Decouples	1	TBD
Stem Feed Piping	Ruptures/fractures Distribution torus	1	TBD

TABLE I-4. (continued)

Component	Failure Mode	Number	Failure Prob/Cycle (x 10E6)
Flow Baffle (SPT Plate and Plena)	Ruptures/fractures	1	TBD
	Decouples	1	TBD
	Buckles	1	TBD
Internal Shield Support	Mechanical failure	1	TBD
Reflector Aluminum Structure	Mechanical failure	1	TBD
	Loss of insulation		
Reflector Peripheral Cylinder	Mechanical failure	1	TBD
One Axial Seal Ring	Fails in place	1	TBD
	Mechanical failure fragments do not remain in place	1	TBD
	Mechanical failure fragments ejected		
One Plunger Assembly	Fails in place	1	TBD
	Mechanical failure fragments do not remain in place	1	TBD
One Attachment Bolt (Reflector Assembly)	Mechanical failure	1	TBD
One Actuator (CDOA)	Excessive external leakage	1	TBD
	Total	27	TBD

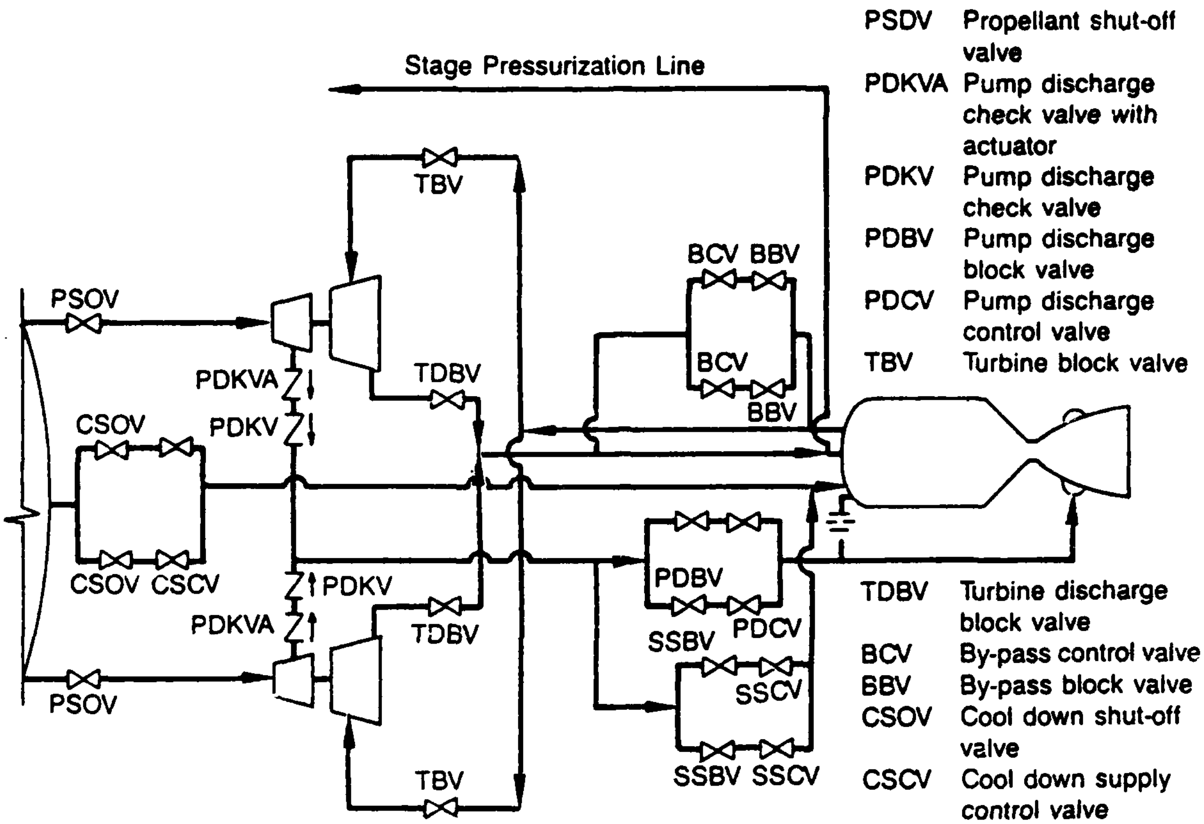

Figure I-2. NERVA engine schematic.

b. Category IIIB: Failures that require Emergency Mode Operation.

2. Category IV: Failures that directly injure the crew, endanger the Earth's population, or damage the spacecraft or other stage modules upon which crew survival depends, and/or that preclude Emergency Mode Operation. This category includes failures that produce one or more of the following system effects:

    a. Uncorrectable thrust vector misalignment

    b. Loss of thrust to less than that required for Emergency Mode Operation

    c. Inability to reduce thrust or unsuccessful shutdown and/or cooldown that precludes engine restart

    d. Unsuccessful startup to attain thrust equal to or greater than that required for Emergency Mode Operation.

Although the NERVA arrangement is slightly different from the current Advanced Nuclear Rocket Engine design (more valves and two turbopumps instead of one), there are enough similarities for some conclusions to be drawn:

1. Elevation of Categories. Since the current ANRE design has only one turbopump instead of the two used by NERVA, those failures listed as Category IIIA become Categories IIIB or IV for the Advanced Nuclear Rocket Engine.

2. Large Probability of Failures. Using Mission Model C of the Task 1 report,[3] the tug would be expected to transport approximately 6600 payloads. A cycle, as used in Tables I-1 through I-4, is defined as a reactor startup and shutdown.[4] Assuming four burns per mission, this equates to a total of 26,400 cycles. From Table I-3, the upper bound for the probability of a

nonnuclear Category IV failure occurring per cycle is about 1.3 per thousand. This is only an upper bound since the events are not mutually exclusive. Taking the component with the largest failure probability, Item 14, a lower bound is seen to be about 0.45 per thousand. Over the life of a tug, the probability of at least one nonnuclear Category IV failure is greater than 0.99 (assuming 26,400 cycles). Figure I-3 is a plot of the probability bounds for a Category IV failure (nonnuclear subsystems) versus number of missions. Although probabilities for nuclear subsystems were not available (Table 4), it seems reasonable to assume that, over the life of the tug, they are also large.

These high failure probabilities, however, are not restricted to the Advanced Nuclear Rocket Engine. The following Phase III (at separation from the Shuttle) failure probabilities are for the Centaur chemical stage.[5]

System	Failure Prob X 10E6
Structures	44
Propulsion	108
Thrust Vector Control	61
Reaction Control System	38
Fluid Systems	28
Avionics	4347
Total	4626

Serious failures, therefore, can be expected to happen. Appropriate actions must be taken so that the consequences are acceptable.

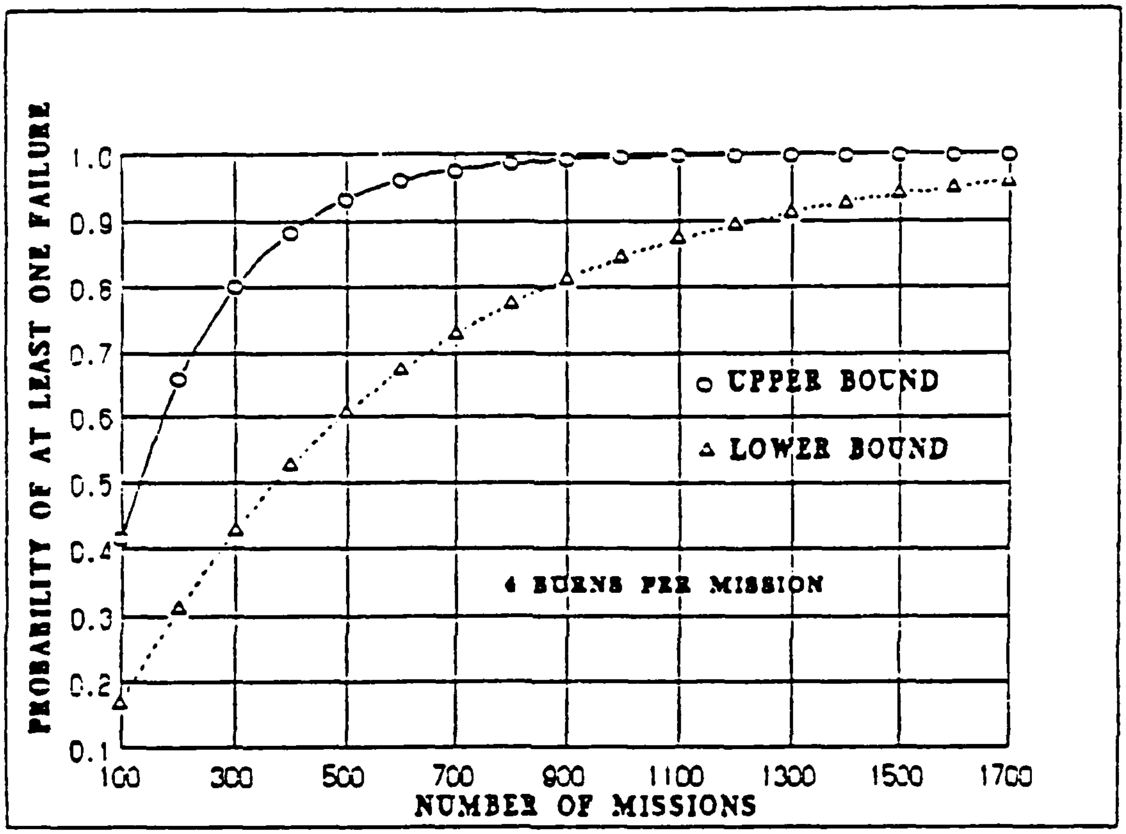

Figure I-3. Probability of Category IV failure vs. number of missions (nonnuclear subsystems).

I.3 ACCIDENT SCENARIOS AND MITIGATION APPROACHES

This section addresses failures that could occur within the major system components and results in one of the adverse situations identified at the beginning of this appendix. Mitigating strategies are also proposed. For each situation, a summary of failure elements is presented first in tabular form and then by detailed commentary.

Some of the scenarios and consequences are from the NERVA flight safety analyses conducted by the Aerojet Nuclear Systems Company and the Westinghouse Electric Company.[2,6,7]

I3.1 Contamination of the Biosphere

This could be caused by any situation which results in radioactive or hazardous material accidentally reentering the biosphere. It could result from unplanned reentry of the tug (or contaminated parts) or contaminated portions of the service platform or spacecraft. Elements whose failure could contribute to the possibility of reentry are summarized in Table I-5 along with mitigation procedures.

I3.1.1 <u>Central Control</u>

1. Surveillance. A failure to correctly locate the various elements could lead to reentry. For example, this might occur if the rendezvous point was incorrectly calculated at a much lower altitude. If there was a failure to monitor the activities of the spacecraft (e.g., deploying an appendage) the results could be damage of the tug, a loss of control, and subsequent reentry. Also, problems could arise if communications were lost with the orbital traffic control center; another craft, for example, could be maneuvering on an intercept course.

2. Communication and Control. Problems here result from an inability to properly control the elements or an inability to

TABLE I-5. CONTAMINATION OF THE BIOSPHERE

Failure Element	Mitigation
Central Control	o Independent Confirmation of Data o Backup Control o Automatic Interrupts o Destruct Mechanism
Spacecraft	o Control Overrides o Sensors for Critical Items o Backup Sensors o Safety Requirements Review
Service Platform	o Control Overrides o Sensors for Critical Items o Backup Sensors o Safety Requirements Review o Controlled Locations for Explosive Components
Tug	o Reduction of Inherent Failure Probabilities o Sensors for Critical Items o Backup Sensors and Communication Equipment o Preprogrammed Computer o Automatic Shutdown o Design/Location of Sensors and Control Equipment o Orbits that do not Intersect the Earth o Destruct Mechanisms

respond to problems detected by the sensors. If communication is lost on an inbound approach, the tug could adopt a reentry course.

3. Mitigation

a. Independent Confirmation of Data. Many catastrophic failures could be avoided by requiring that critical data be verified by an independent source. Such data would include location, velocity, and status of components that could potentially interfere with the mission. If data concerning critical items differed between central control and its backup, the operations would be terminated until the problem was resolved. A logical choice for the independent source would be the orbital traffic control center.

b. Backup Control. In the event of a loss of control by central control, procedures should be implemented to automatically switch to a backup station such as the orbital traffic control center.

c. Automatic Interrupts. In the event communications are lost with the orbital traffic control center, a mission interrupt should be enacted. Depending on the situation, this may involve reactor shutdown and other preprogrammed procedures.

d. Destruct Mechanism. A destruct mechanism should be available (e.g., a fast intercept missile system located at the central control) for use in the event of an uncorrectable emergency. For example, if the tug or other contaminated element were on a reentry trajectory and all attempts to correct the problem failed, the missile would be used to destroy the element.

I.3.1.2 Spacecraft

1. Erratic Behavior (deploying an appendage during docking), communications that interfere with control, inadequate knowledge concerning the status of key components, or commands received from another control stations could cause a collision and subsequent inability to guide the tug.

2. Mitigation

   a. Control Overrides. A procedure to ensure absolute control of all operations by central control is essential. This could include overrides so that the normal control headquarters cannot interfere and a positive shutdown of potentially dangerous elements.

b. Sensors for Critical Items. Central control must have knowledge of the status of all components that could potentially interfere with the mission. As a minimum, the status of the following items should be monitored by sensors: status of appendages, temperatures, orientation, velocity, and altitude.

c. Backup Sensors. Backups for the sensors mentioned above should be available.

d. Safety Requirements Review. Additionally, the spacecraft must be required to undergo a detailed safety requirements review as part of its final checkout procedures.

I3.1.3 Service Platform

Same as above for the spacecraft. Additionally, explosive components such as hydrogen and oxygen should not be stored in close proximity to each other.

I3.1.4 Tug

1. The possibility of unplanned reentry would be increased by problems in any of the subsystems covered below.

   a. Gimbal Assembly. The gimbal assembly subsystem provides guidance control for the vehicle, and as such, is critical to mission success. Failures in this area can have catastrophic consequences if no corrective actions are provided. For example, the gimbal pivot could fail (structural failure of the yoke or a trunnion). The probable result would be a loss of shutdown/cooldown capability with subsequent disassembly of the reactor. Also, the gimbal actuators could fail in place. This would result in loss of ability to respond to vehicle directional

commands. Tumbling, hardover, and vehicle breakup are possibilities.

b. Reactor Assembly. The engine has internal shielding in the pressure vessel to protect various components. A failure here could cause problems in a number of areas, including the gimbal assembly and the communications and status sensors.

c. Performance Sensors, Control and Communication Equipment. Equipment must be available for real-time monitoring of various components to relay these data to the control station and to take any required actions.

d. Nozzle, Pressure Vessel, and Closure Assembly. An external leakage from the exhaust gas stream (at the seal between the nozzle and nozzle extension) would result in a thrust reduction and misalignment. A structural failure of the pressure vessel would be catastrophic to the engine with resulting disassembly of the reactor.

e. Thrust Structure and External Shield. Consequences of a failure of the thrust structure to transmit thrust along the proper axis or failure to support interface components could range from thrust misalignment to catastrophic structural collapse with rupture of critical engine components. Reactor melt and disassembly is a possibility. A structural failure of the external shield could cause damage to other critical engine components such as propellant lines. This could cause possible loss of coolant supply capability leading to subsequent reactor disassembly.

2. Mitigation

a. Reduction of Inherent Failure Probabilities. Possibilities for accomplishing this include incorporating redundancy, developing better designs, and using stronger materials. The impact will be small, however, considering the large number of missions contemplated.

b. Sensors for Critical Items. The first line of defense against a major failure is adequate knowledge of the situation as it is developing. Items whose performance could have serious safety implications must be carefully monitored. Items that must be monitored include radiation levels, fuel temperatures, control reflector positions, reactor power level and rate of change, status of any appendages, orientation, altitude, and velocity.

c. Backup Sensors and Communication Equipment. As above, these are essential.

d. Preprogrammed Computer. This could act as a backup in case of lost communications. Depending on the situation, it may be programmed to shut down the reactor and take certain other actions in the event communications are lost.

e. Automatic Shutdown. Controls should be set for reactor shutdown if certain dangerous events occur; for example, loss of ability to control excessively high temperatures.

f. Design/Location of Sensors and Control Equipment. Sensors and controls should be carefully designed and located so that the possibility of damage is minimized. For example, they could be located above the propellant tank to minimize concerns associated with loss of the shield or impact damage. An advantage of this location--in addition to the

distance involved--is the neutron attenuation provided by the hydrogen.

g. Orbits That Do Not Intersect the Earth. Orbits that intersect or come within close proximity to the Earth must be avoided; this will preclude some of the problems associated with a thrust control failure.

h. Destruct Mechanisms. This is the final line of defense in the event of a total loss of ability to control the situation. Destruction could be accomplished through either an onboard or external device.

I3.2 Contamination of the Service Platform

This would be caused by the tug and could occur while it is attached (for maintenance, etc.), during disengagement, or during docking. The contamination could be caused by release of radioactive material or other hazardous substances such as beryllium. Additionally, since it is assumed that the platform would be manned, the safe radiation limits may be exceeded. Possible contributing elements, along with mitigating procedures, are summarized in Table I-6.

I3.2.1 Central Control

1. Problems could result from a failure to properly determine the location of the platform and tug, which leads to an inability to compute the correct intercept point and results in a collision. This could be caused by faulty or inoperable sensors or computers. Additionally, interference could be caused by incomplete information concerning other activities. For example, another craft could be approaching or taking off in the path of the tug.

TABLE I-6. CONTAMINATION OF THE SERVICE PLATFORM

Failure Element	Mitigation
Central Control	o Transfer of Control o Operational Interrupts
Tug	
During Return/Attachment	o Nonintersecting Orbits o Reactor Shutdown o Control Transferred to Platform
While Attached	o Depressurization
During Departure	o Use of Robot
Service Platform	o Sensors for Critical Items o Backup Sensors o Controlled Location for Explosive Components o Separate Shielded Area

2. Mitigation

a. Transfer of Control. The platform would be expected to have real-time knowledge concerning events in its surroundings. One approach, therefore, would be to stop the tug at a certain distance from the platform. A transfer of control would then be made and the platform would control final approach and docking. Central control would change to a backup status.

b. Operational Interrupts. As before, if communication is lost with traffic control, the operation would be interrupted.

13.2.2 Tug

1. During return attachment. As described above, a number of problems could arise that would result in a loss of ability to control the tug. If the tug were on a return orbit that

intersected or came close to the platform and control were lost, a collision could result. Problems could also arise on attachment because of inherent difficulties involved in precise maneuvering of such a relatively large object.

2. While attached. A damaged shield that was not detected could be a serious hazard. An unsafe condition could arise because of problems within the reactor (assuming it is not shut down and checked prior to final approach), an accidental collision, or a maintenance error. Additionally, a faulty valve, a break in the lines, or a puncture of the storage tank could cause problems.

3. During departure. An accident could occur as a result of loss of control mechanisms.

4. Mitigation

   a. Nonintersecting Orbits. Orbits must be selected for return that do not intersect or come dangerously close to the service platform.

   b. Reactor Shutdown. A procedure to mitigate the consequences of many of these failures would be to shut down the tug's nuclear engines at a safe distance from the platform. The safety of all components would be verified before final approach was approved.

   c. Depressurization. In order to avoid the problems caused by leaks or punctures of the hydrogen tank, the system should be depressurized at the transfer point.

   d. Control Transferred to Platform. Same as above.

   e. Use of Robot. Because of its ability to maneuver, the robot could be used to provide a more detailed safety check of the tug prior to final approach. This could be accomplished

using onboard sensors and through visual observations. Also, since it could be easier to maneuver, the tug would be used for final approach. The robot would remain attached to the tug during maintenance. In the event an emergency developed, the robot could be used to rapidly remove the tug to a safe distance. It would also be used for normal departure; at a safe distance, the tug's nuclear engines would be started and control transferred to central control.

I3.2.3 Service Platform

1. Problems could occur as a result of the platform's deploying an appendage, transmitting on frequencies that interfered with control, providing inadequate facilities, storing explosive materials improperly, or a number of other actions that interfered with the mission.

2. Mitigation.

    a. Sensors for Critical Items.

    b. Backup Sensors.

    c. Controlled Location for Explosive Components.

    d. Separate Shielded Area. *The maintenance area should be* located away from the most populated areas of the platform. Additionally, it should provide radiation shielding (in the event of a failure of the tug's shielding) and protection from micrometeoroids and accidental collisions. The work environment must be closely monitored.

I3.3 Contamination of the Spacecraft

This could be caused either by the tug colliding with the spacecraft or by problems that develop on the tug while it is in close proximity with

the spacecraft. Possible scenarios are noted in Table I-7 and discussed below.

TABLE I-7. CONTAMINATION OF THE SPACECRAFT

Failure Element	Mitigation
Central Control	o Independent Confirmation of Data o Automatic Interrupts o Backup Control
Spacecraft	o Control Overrides o Sensors for Critical Items o Backup Sensors o Safety Requirements Review
Tug	o Reduction of Inherent Failure Probabilities o Use of Robot

I3.3.1 Central Control

1. An error by central control is the easiest way to effect a collision and subsequent contamination. These failures were covered under contamination of the biosphere and are summarized in Table I-7.

2. Mitigation. Covered under contamination of the biosphere and summarized in Table I-7.

I3.3.2 Spacecraft. Failure modes and mitigation procedures were covered under contamination of the biosphere and are summarized above.

I3.3.3 Tug

1. Contamination could be caused by problems in those elements identified previously (gimbal assembly, pressure vessel, and thrust structure) that lead to disassembly of the core. Also, a

collision could be caused as a result of the inherent difficulties involved in maneuvering the relatively large tug.

2. Mitigation. Since it would be easier to maneuver the robot, the possibility of problems during docking could be developed so that if a potentially dangerous situation is developing in the tug, immediate disengagement--using the robot--would be effected.

I3.4 Loss of Engine Capabilities

The engine is considered to be lost when it suffers damage so extensive that it would have to be replaced. Applicable failures and mitigation approaches are noted in Table I-8.

I3.4.1 Category III, IV Failures

1. Many of the failures listed in Tables I-1 through I-4 have the potential to seriously damage the engine unless the developing situation is detected rapidly and appropriate action taken.

2. Mitigation

   a. Reduction of Inherent Failure Probabilities.

   b. Sensors and Backup Sensors For Critical Items. In order to avoid major damage to the engine, action would have to be taken within seconds of the failure for many of the items. This requires careful consideration of the detectors needed and provision for backup.

   c. Preprogrammed Computers. In most cases, the first step would be reactor shutdown. In order to save time and as a backup in case of communication difficulties, an onboard computer could be programmed to shut down the reactor in certain dangerous situations.

TABLE I-8. LOSS OF ENGINE CAPABILITIES

Failure Element	Mitigation
Category III, IV Failures	o Sensors and Backup Sensors for Critical Items o Preprogrammed Computers o Robot
Collisions	
Spacecraft	o Table I-5 Table I-6
Service Platform	o Table I-5 Table I-6

d. Robot. The robot may serve a useful function in such emergencies. Because of its ability to detach from the tug and maneuver, it may be able to perform additional surveillance in order to define more precisely the situation for the control station. Additionally, it could be equipped to perform certain types of repair and adjustments. In any case, it could be used to detach and move the spacecraft to a safe distance.

I3.4.2 Collisions

These were covered in Tables I-5 and I-6.

I3.5 Loss of Mission

The mission of the tug and the other elements is to transfer the spacecraft from one orbit to another. If the tug or other elements (excluding the spacecraft) are destroyed or rendered inoperable, the mission can be accomplished by merely replacing them unless the mission is time dependent. In that case, these losses would possibly result in loss of the mission. Neglecting the time dependence, the only way the mission can be lost is to lose the spacecraft. It can be lost either by destruction or by suffering damage so severe that it would have to be replaced. Table I-9 provides a summary of factors affecting loss of mission.

TABLE I-9. LOSS OF MISSION

Failure Element	Mitigation
Central Control	o Independent Confirmation of Data o Automatic Interrupts o Backup Control
Spacecraft	o Control Overrides o Sensors for Critical Items o Backup Sensors o Safety Requirements Review
Tug	o Reduction of Inherent Failure Probabilities o Sensors for Critical Items o Backup Sensors and Communication o Preprogrammed Computer o Automatic Shutdown o Use of Robot

1. Loss of the spacecraft can be accomplished either by collision with the tug or by a problem that developed within the tug while it was in close proximity with the spacecraft.

2. Mitigation. Procedures were covered in Tables I-5 and I-6 and are summarized above.

I.4 SUMMARY/CONCLUSIONS

This appendix addresses safety issues concerning the use of a nuclear rocket in Earth orbit transfer missions. It identifies the major adverse situations that could occur, describes typical transfer missions, and explains the roles of participating elements. The magnitude of the potential problem is illustrated through a summary and analyses of the results in an earlier safety study of the NERVA engine. Finally, failure modes, consequences, and mitigating procedures are identified for each adverse situation.

The following conclusions are drawn:

1. The probability of a serious failure for each operation of the nuclear rocket (one startup and shutdown) is low, i.e., between 0.5 and 1.3 per thousand. For the projected number of missions, however, this probability approaches 1.0.

2. Based upon the various failures, mitigating strategies have been identified to protect against contamination of the biosphere and service platform. The following actions would be required:

   (a) Provide centralized control of the operation. Since each of the participating elements can contribute to a catastrophic failure, the entire operation must be centrally orchestrated. The central control station must have knowledge of the total space environment and must have a backup facility that is not co-located with central control. The backup would be used in an emergency and would also provide independent confirmation of critical data. Control overrides (along with redundant and carefully placed sensors, communication, and control equipment for each element) would be required.

   b. Implement operational procedures that emphasize safety. As with design of the system, development of operational procedures must address safety issues from the beginning. A number of procedures can be implemented that would lead to a safer environment; these procedures include the following: transferring control to the platform when the tug is operating nearby, using the robot for precise maneuvering, using orbits that do not intersect or come within close proximity to the Earth or service platform, providing a separate and shielded area for the tug on the platform, and providing separate storage locations for potentially explosive components.

c. Incorporate equipment and methodologies to mitigate the consequences after a failure has occurred. Failures will occur and mitigating strategies and equipment must be incorporated in the design process. Programmed computers on the tug are a possibility. In the event an emergency developed, corrective actions could be taken, e.g., shutting off the reactor. Operational interrupts should be initiated in the event communications are lost or there is a conflict in the critical data received. In the event all corrective measures fail, a destruct mechanism should be available. This could be on the tug or in a fast reaction missile system.

3. The possibilities of contamination of the spacecraft, loss of engine, and loss of mission can also be reduced, but not completely mitigated, by the actions identified above.

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APPENDIX J

POWER PREDICTION FOR NUCLEAR ROCKET TYPE FUEL ELEMENT IN ATR

EG&G Idaho

INTEROFFICE CORRESPONDENCE

Date: April 13, 1987

To: Z. R. Martinson

From: B. L. Rushton

Subject: POWER PREDICTION FOR NUCLEAR ROCKET TYPE FUEL ELEMENT IN ATR CORNER LOBE - Rush-3-87

The calculations that you requested to examine the feasibility of using the ATR or an equivalent type reactor for testing nuclear rocket fuel have been completed. The purpose of the study was to determine if .9 MW of power could be generated in the composite fuel matrix with the ATR lobe operating at the 60 MW power level. This task was accomplished using the basic one-dimensional corner lobe model developed for ATR neutronics analyses. This geometric model was modified to represent the rocket fuel test module in the SCAMP S_n transport theory code. The results show that under the specified operating conditions about 1.0 MW of test power would be attained. Calculational details and results are given in the following paragraphs.

The basic one-dimensional model used for ATR corner-lobe calculations is shown in Table 1. This is a coarse region model description. The actual working model as represented in the transport theory calculation has each fine region represented discretely which results in a model having 44 regions. This 44-region model was modified for the current study to represent the single rocket fuel test module. Coarse regions one and two of Table 1 were modified as shown in Table 2. From the outer radius of 2.38125 to the outer boundary, the basic corner-lobe model was not changed.

Nuclide atom densities for the test fuel module were determined using the information that you provided. Namely, the figures showing full length and plan views of the composite fuel element and the table having chemical analysis data for the fuel matrix. The highest fuel loading of 598 kg U/m^3 was used in the analysis with an assumed enrichment of 93%. Using these data, the rocket fuel module was homogenized on a volume fraction basis into a single composition having the nuclide atom densities given in Table 3. The corresponding volume fractions are also included.

TABLE 1. ONE-DIMENSIONAL MODEL

Region	Description	Outer Radius (cm)
1	Experiment specimen and holder	2.04470
2	Shroud (Zr), water and flow tube	2.38125
3	Water pressure tube, He, and insulation jacket	3.65125
4	Water and safety rod guide tube	4.76250
5	Water and flux trap filler	6.35000
6	Water and flux trap baffle	7.38505
7	Water	7.56260
8	Fuel plate 1	7.96392
9	Fuel plate 2	8.28904
10	Fuel plate 3	8.61416
11	Fuel plate 4	8.93928
12	Fuel plate 5	9.26440
13	Fuel plate 6	9.58952
14	Fuel plate 7	9.91464
15	Fuel plate 8	10.23976
16	Fuel plate 9	10.56488
17	Fuel plate 10	10.89000
18	Fuel plate 11	11.21512
19	Fuel plate 12	11.54024
20	Fuel plate 13	11.86536
21	Fuel plate 14	12.19048
22	Fuel plate 15	12.51560
23	Fuel plate 16	12.84072
24	Fuel plate 17	13.16584
25	Fuel plate 18	13.49096
26	Fuel plate 19	13.94308
27	Beryllium-water annulus	14.40739
28	Pure beryllium Region A	16.67612
29	Intermediate beryllium reflector	22.19808
30	Beryllium-water Region C	24.10308
31	Hf drum at 4" from fuel	24.73808
32	Beryllium-water Region C	31.72308
33	Core tank	35.56000
34	Water (125F)	81.28000

Cross Sections

The basic 31-group ATR library is based on ENDF-4 data and is a special application library that does not have values for the fission cross sections. This library also does not include data for the H_2 coolant in the rocket fuel module. When using this library in a SCAMP calculation to compute power density, one has to compute power from the neutron source and normalize the resulting power to the desired power level by hand calculation. That is, SCAMP requires fission cross sections for output power normalization. To avoid the H_2 omission and hand normalization, the ATR fuel plates and the rocket fuel test were represented with cross sections derived with the COMBINE-5 spectrum code and the latest ENDF-5 based cross section library. This required derivation of six macroscopic cross-section sets. The ATR library was then updated to include the new macroscopic data sets. As seen in the results section, both libraries were subsequently used to represent fuel in the SCAMP calculations. This provided a convenient method for validation of the new macroscopic data.

The power density result from the normalized (60 MW) SCAMP calculation gives the watts per cm^3 of homogenized test fuel for a 1.0 cm thick slice through the average plane of the axial distribution. The corresponding watts per cm^3 of the composite fuel matrix is then obtained by dividing the SCAMP result by the composite volume fraction (.68819). The total composite test power is then computed by multiplying the resulting watts per cm^3 by the composite matrix volume and for these cases the ATR height was assumed as opposed to the 132 cm length of the rocket fuel module.

The one-dimensional transport results are summarized in Table 4.0. Results from two cases are given to show the consistency obtained using two different sets of cross sections to represent the fueled regions of the one-lobe model. One case has all fuel constants calculated with COMBINE-5 using the latest version of ENDF-5 basic cross-section data. The other case used the basic ATR 31-group set of cross-section data that are ENDFB-4 based. When using this library, the H_2 test coolant was not included due to its omission from the library. Essentially the same test power is produced in the SCAMP calculation when using these different libraries. The observed difference in eigenvalue is primarily from two sources. One is version 4 versus version 5 ENDFB data and the other is caused by the thermal group's disadvantage factors. These factors were assumed to be unity for the ENDFB-5 library generation.

An additional ancillary computation was made to investigate the power production attainable in the large test bundle that has six composite fuel modules surrounding a tie-rod support module. This large test would occupy most of the space internal to the stainless steel pressure tube of the corner lobe. The required power for such a test was 5.5 MW with the ATR lobe power at 60 MW. The ancillary computation result was only 40% of the requirement. Material changes in the central tie-rod region such as a full hexagonal element composed of beryllium might provide some increase; however, this was not pursued in the current study.

ds

Attachments:
As Stated

cc: J. D. Abrashoff
A. W. Brown
J. A. Lake
J. H. Ramsthaler
B. L. Rushton File (2)

TABLE 4. SINGLE ROCKET FUEL TEST MODULE POWER (ATR LOBE AT 60 MW)

Library	Eigenvalue	Power (MW)[a]
ENDF-5	1.1341	1.01
ENDFB-4	1.1178	1.05

a. The test power is for a test module length equal to the active height of the ATR fuel element (121.9 cm).

3